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THE DESIGN AND DEVELOPMENT OF A SHORT-DURATION CONSTANT PRESSURE COMBUSTOR FOR USE IN ROCKET BASE HEATING INVESTIGATIONS

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ABSTRACT

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The design philosophy and development program for short-duration combustors to simulate liquid-rocket engine exhaust flows is reported. This particular combustor has been employed to duplicate liquid hydrogen-liquid oxygen and RP-lox systems, but is applicable to any liquid system for which the fuel and oxidizer can be adequately synthesized by gases or mixtures of gases. The basic principle is to store the fuel and oxidizer gases in charge tubes separated from a mixer-injector and combustion chamber by diaphragms. Rupture of the diaphragms permits the gases to flow into the combustor whereupon ignition and burning takes place. During the length of time required for expansion waves to travel the length of the charge tubes and return, supply conditions are steady.

The design considerations for each of the system components are discussed. Operating characteristics and performance data are presented.

The primary advantages of the short-duration approach are that scale-model testing may be done by an economical and safe method that avoids the complex engineering and operational problems associated with the fueling, combustion, ignition and required cooling of conventional continuous-flow engines; that high-altitude simulation is practical with a very modest vacuum pumping system; that reliable instrumentation techniques, particularly heat transfer and pressure, which have been developed for hypersonic shock tunnel testing may be employed; and that the technique is readily adaptable to a wide variety of parametric studies. AUTHOR

FOREWORD

The research and development reported herein was conducted for the George C. Marshall Space Flight Center, NASA, under Contract NAS 8-823. This program is under the cognizance of the Aeroballistics Division of MSFC with Messrs. W. K. Dahm and H. B. Wilson, Jr., as Technical Supervisors.

The program concerned with the application of short-duration techniques to the experimental study of rocket base heating was started in December 1960. This report deals with only one phase of the overall program--the development of a suitable short-duration combustor for conducting scale model experiments of liquid rocket systems. Other phases of the overall program have been and will be reported separately.

The authors wish to acknowledge the efforts of their colleagues at CAL for their contributions to this program, particularly to Mr. R. C. Weatherston for his original conception of the constant pressure combustor and his advice and assistance during the development program, to Mr. K. D. Bird who proposed the final hybrid combustion configuration, and to Messrs. J. W. Reece and R. J. Dennis who participated extensively in the actual experimental program.

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SECTION I INTRODUCTION

Rocket base heating and flow recirculation studies in support of the NASA/MSFC Saturn program are in progress at CAL. The experimental studies are based on the concept that rocket engine flows need be duplicated in the laboratory only for a length of time sufficient to establish steady flow patterns and make the desired measurements. Where the flow establishment processes are governed mainly by wave phenomena, it has been shown that base flows may be established in the order of 1 or 2 milliseconds and that shock tube techniques are directly applicable to rocket flow studies.[1,2,3]

The primary advantages of the short-duration techniques are that scale model testing may be done avoiding the many engineering and operational problems associated with the fueling, combustion, igntion and cooling of conventional continuous flow engines; that high-altitude simulation becomes practical with very small vacuum pumps; and instrumentation, particularly heat transfer,[4] and pressure,[5] which has been developed for hypersonic shock tunnel testing may be employed.

An important element of the short-duration approach is, of course, the short-duration combustor used to generate the propulsive gases. Ref. 2 describes a rudimentary isochoric (constant volume) hydrogen-oxygen combustor which was essentially a combustion shock tube driver. Although the performance of this combustor was satisfactory, it appeared that other techniques might be used which preserved all of the desirable features of the isochoric combustor, and at the same time offered such improvements as maintenance of constant reservoir conditions, longer test times and elimination of multiple spark plugs. This generic combustor is based upon isopiestic or constant pressure combustion.

The combustor program discussed in this report applies specifically to liquid hydrogen-liquid oxygen stages with which the early phases of the CAL program were primarily concerned; however, the techniques are generally applicable to any rocket engine whose combustion products can be synthesized with the combustion of appropriate gases. In yet unreported studies at CAL, they have been used to simulate a RP-lox system by the use of ethylene and oxygen.

SECTION II CONSTANT VOLUME COMBUSTOR

Fig. 1 is a wave diagram of the constant volume combustor employed in the studies reported in Refs. 2 and 3. A shock tube driver was charged with a hydrogen-oxygen mixture. Combustion of this mixture was then accomplished by the simultaneous firing of a series of spark plugs located every four inches along the length of the shock tube. The combustion raised the pressure within the tube to the desired operating pressure, or slightly above, which was sufficient to rupture the diaphragm at the downstream end of the tube and permit the combusted gases to flow into a small plenum chamber and out the nozzles as indicated schematically in the figure. The flow of gas sets up a centered non-steady expansion wave at the downstream end of the shock tube. During the length of time required for the head of the expansion fan to be propagated to the far end of the tube and be reflected back to the plenum, the reservoir conditions, pressure and temperature are constant if there is no heat loss from the gas to the shock tube walls. The wave propagation speed is determined by the acoustic properties of the gas. For the hydrogen-oxygen combustion, a tube 15 feet long provided about 4.5 milliseconds of test time. Fig. 2 is a photograph of the actual apparatus.

In practice, the heat transfer to the shock tube walls is not negligible and the reservoir conditions do not remain constant. Fig. 3 is a plot of the measured reservoir pressure as a function of time. It may be seen that the pressure reaches the maximum value in about 0.5 milliseconds. This is followed by a gradual decrease which amounts to an average of 5 percent per millisecond over the 4.5 milliseconds prior to the arrival of the expansion.

By assuming that the base flow processes are quasi-steady, a timewise resolution of the combustor pressure may be made to identify the instantaneous reservoir conditions. Although this procedure appeared to be justified, it was considered desirable to eliminate this disadvantage of the constant volume combustor if possible. Accordingly, development of a constant pressure device was initiated as described in subsequent sections.

SECTION III CONSTANT PRESSURE COMBUSTOR

A. Premixed Gases

The principles of operation of a constant pressure combustor using premixed fuel and oxidizer gases may be described by considering the schematic and wave diagrams of Figs. 4 and 5. The premixed gases are loaded in the gas supply tube which is separated from a combustion chamber at its downstream end by a diaphragm. Mechanical rupture of this diaphragm permits flow of the mixture into the combustion chamber where ignition takes place. As in the case of the constant volume combustor, supply conditions are constant during the time of transit of the head of the expansion fan up the tube and back. Since the gas mixture is now cold, however, the acoustic speed is considerably less and the same length of tube will give substantially longer test time or, conversely, for a given test time a much shorter tube may be used.

A development of such a premixed gas, constant pressure combustor was conducted. A drawing of the configuration is shown in Fig. 6. It was desired that the final combustor simulate either a hydrogen-oxygen engine or an RP-lox engine. The latter was to be done by the use of ethylene (C_2H_4) as the fuel gas and oxygen. The charge tube was loaded through a mixing valve with the proper mixture of fuel and oxidizer. Two mylar diaphragms in series with a small space between them were used. This cavity between the diaphragms was loaded to half the final charge pressure through a solenoid valve connected between this section and the charge tube. When the charge tube was fully loaded, the solenoid valve was opened, admitting full charge pressure between the diaphragms and rupturing first the downstream, and then the upstream diaphragm. The gas flow then proceeded through the metering nozzle and supersonic diffuser forming a standing normal shock wave at the downstream end of the diffuser. Continuing, the flow passed through a perforated steel cone into the combustion region. A glow plug ignited the mixture with the steel cone acting as a flameholder. A longitudinally adjustable spacer with holes in it was provided to simulate the throats of the rocket nozzles and to control the size of the combustion chamber. In this configuration, test time was calculated to be 10 milliseconds for a hydrogen-oxygen mixture and 15 milliseconds for an ethylene-oxygen mixture.

Among the difficulties experienced with this combustor were ignition failure, pre-ignition, and flashback of the combustion into the supply tube. All of these problems were largely overcome, but in so doing much of the desired basic simplicity was lost and reliability was poor. The inset in Fig. 5 shows a pressure time history in both the combustion chamber and in the charge tube. It may be seen that about 15 milliseconds of steady combustion was obtained before flashback occurred. The severity of the resulting detonation wave drove the charge tube pressure gage off scale. A major disadvantage of this combustor was the necessity of dealing with an explosive mixture of gases in the charge tube which presented a constant safety problem. When flashback occurred, conditions were favorable for the development of destructive detonations within the tube. For these reasons, further development on the constant volume combustor with premixed gases was discontinued.

B. Separate Fuel and Oxidizer Gases

The primary difference between the final constant pressure combustor and that described above is that the actual gas mixing is not accomplished until combustion is to take place, mixing being done immediately upstream of the combustion chamber as shown in Fig. 4. The initial combustor shown in Fig. 7 incorporated two separate charge tubes sealed at the ends with mylar diaphragms. A spring plunger actuated a piston to which was connected two cutters. When the mechanism was triggered, the cutters simultaneously opened the diaphragms at the ends of the charge tubes permitting the gases to flow out through the mixer plate into the combustion chamber.

A small shock tube containing a mixture of hydrogen and oxygen was used as an igniter tube. This tube was isolated from the combustion chamber at one end by a mylar diaphragm. A spark plug was located at the opposite end which was fired shortly after the activation of the diaphragm cutters. The ensuing increase in pressure ruptured the mylar diaphragm. The ignited fuel then flowed into the combustion chamber igniting the mixed gases from the charge tubes. Various initial pressures of the igniter tube were used, and the time delay of firing was varied to find the optimum combustor pressure history. The pressure-time record shown in Fig. 8 was the best obtained.

SECTION III (Cont'd.) CONSTANT PRESSURE COMBUSTOR

An initial inflow of gas into the plenum chamber has taken place when the igniter tube flow is initiated. This produces an immediate ignition of gases in the plenum with a sharp rise in pressure. However, this is a transient effect and insufficient gas has flowed into the combustion chamber to completely fill the chamber to the design pressure; also the burned gases start flowing out of the plenum. The net result is a drop-off in pressure from the initial peak. However, as the nozzles become choked, the pressure in the plenum builds up effectively, attaining equilibrium after about 4.5 to 5 milliseconds. From this time until the reflection of the expansion wave in the supply tube, the flow is steady. Thus, if there are no other conditions limiting the test time, the performance of this combustor is satisfactory; and it is necessary only to provide sufficient length of the supply tubes to give the required test time. In conducting investigations in the high-altitude base heating facility,[2, 3] however, reflections from the tank walls do limit the test time and the long starting time is not acceptable.

It may be seen from Fig. 3 that the constant volume combustor possessed very favorable starting characteristics, i.e. fast rise time without large pressure transients, and that its shortcomings were manifest only late in its operation; while the constant pressure combustor configuration described exhibited good performance only after its starting transients had subsided in its operation. These opposite characteristics suggested combining the favorable features of both in a hybrid combustor which is basically a constant pressure device but employs a constant volume start. By placing mylar diaphragms at the rocket nozzle throats or just upstream of them, the combustion chamber may be isolated from the charge tubes, permitting it to be preloaded with a hydrogen-oxygen mixture. The initial pressure is selected such that the pressure after combustion is the operating pressure of the device. Operation in this manner was attempted; and although the starting pressure transients were never completely eliminated, significant improvement was made with the start time being reduced to 2.5 milliseconds. During this phase of the development, however, the other modifications were possible which resulted in a more simple configuration and essentially ideal performance.

SECTION III (Cont'd.) CONSTANT PRESSURE COMBUSTOR

First, combustion chamber loading may be accomplished by simply cutting the charge tube diaphragms and allowing the gases to flow into the combustion chamber. Second, if the combustion chamber is initially pumped to a moderately high vacuum, the gas flowing into the combustion chamber will experience an adiabatic compression which is adequate to raise the mixture to its autoignition temperature; thus, no spark is necessary.* Furthermore, if the ignition occurs almost instantaneously, which it does, no large pressure pulses will be created which blow out the exit nozzle diaphragms. Hence, these diaphragms can be sized to burst at or just below the equilibrium pressure. The process is one then in which the main charge tube diaphragms are cut, and the mixed gases autoignite and burn in the combustion chamber at ever increasing pressure until the operating pressure is reached. At this point, the nozzle diaphragms burst and the rocket nozzle flows commence. One further simplification is also made. Since the rocket nozzles typically exhaust into an altitude chamber at low pressure, minute holes may be pierced in the nozzle diaphragms so that the altitude chamber itself evacuates the combustion chamber.

Fig. 9 shows a typical combustor pressure record from the final configuration. On this run, a pitot probe was located at the exit of one of the nozzles. It may be seen that there is no output from this pitot probe until the full combustor pressure is reached. At this point, there is a sharp step in pressure followed by a period of steady flow. The final hybrid combustor has evolved into a very simple device consisting of two charge tubes, a mixing plate, and combustion chamber. Photographs of the various components are shown in Fig. 10.

*If autoignition is delayed for any reason, large overpressures could be experienced. Therefore, a spark plug is normally used as a precautionary measure.

SECTION IV COMBUSTOR DESIGN METHODS

This section discusses the various criteria from which the constant pressure combustor was designed.

A. Fuel and Oxidizer Supply Tubes

The length of the charge tubes is governed by the desired test time and the gases to be used. The tube diameter determines the maximum allowable mass flow and has an effect on the test time.

1. Supply Tube Length

As the diaphragm of the charge tube is cut, the gas adjacent to the orifice is accelerated by an expansion wave, as shown in Fig. 11, causing the gas to flow out of the supply tubes. The head of the expansion wave propagates with the velocity of sound into the stagnant gas. The pressure in Region 2 remains constant until the expansion wave returns from the closed end of the charge tube. For a small disturbance, the time, t , that the tube acts as a constant pressure reservoir is given by the following relationship:

$$t = 2\ell/a \qquad (1)^{*}$$

where ℓ is the length of the tube and a is the velocity of sound in the gas in the tube. Fig. 12 shows the time as a function of tube length for various gases. Hydrogen with its low molecular weight, and therefore high acoustic velocity, imposes the most severe requirements. Even with hydrogen, a tube which is only 10 feet long will give nearly 5 milliseconds of steady flow time.

The head of the reflected expansion wave is accelerated by the tail of the incident expansion wave; thus the strength of this wave influences slightly the overall test time (see Fig. 13). The pressures in Regions 1 and 2 are related by the Mach number in the supply tubes as shown below:[6]

$$\frac{p_1}{p_2} = \left(1 + \frac{\gamma - 1}{2} M\right)^{\frac{2\gamma}{\gamma - 1}} \qquad (2)$$

p_2 is the effective static pressure of the gas upstream of the orifice.

*All symbols are defined on page 16.

2. Supply Tube Diameter

In order to provide control of the mass flow into the combustion chamber, orifices are used in the supply tubes just upstream of the diaphragms. If the pressure ratio across the orifice is sufficient to choke the flow through the orifice, then from continuity

$$d_t^2 D_2 = d_e^2 D_* \qquad \text{(see Fig. 14)}$$

or

$$D_2 = (d_e/d_t)^2 D_* \qquad \text{"} \qquad (3)$$

Therefore, the Mach number in the supply tube is only a function of the ratio of the effective orifice diameter to the supply tube diameter (d_e/d_t), as long as the orifice is choked (Fig. 15). Since increased Mach number of the flow in the supply tubes decreases the testing times, it is desirable to keep (d_e/d_t) small, preferably below 0.4.

If the flow through the orifice becomes subcritical, Eq. 3 should be rewritten as:

$$D_2 = (d_e/d_t)(D/D_*)\, D_* \qquad (3a)$$

where (D/D_*) is dependent on the pressure ratio across the orifice (see Fig. 16). In determining the pressure ratio, it is assumed that the chamber pressure is acting downstream of the orifice. This assures that the static pressure recovery is equal to the static pressure loss through the mixer plate. This assumption has given satisfactory results for the present constant pressure combustor and is probably adequate for most design purposes. The orifice pressure ratio is then: $r = \frac{P_{cc}}{P^\circ_2}$

B. Orifice Selection

The charge tube orifice diameters have to be chosen so that the mass outflow through these orifices is exactly equal to the mass outflow through the nozzles of the model while maintaining the proper oxidant-fuel mixture ratio. The weight flow through the orifice can be expressed as:

$$\dot{W} = \left(\frac{\gamma g}{R}\right)^{1/2} \frac{p_2^\circ \alpha_e D_1}{(T^\circ)^{1/2}} \qquad (4)$$

SECTION IV (Cont'd.) COMBUSTOR DESIGN METHODS

Parametric solutions for this equation are presented in Fig. 17.

This weight flow is independent of the pressure downstream of the orifice as long as it is choked. When the flow through the orifice is subcritical, the mass flow becomes dependent on the pressure ratio across the orifice and Eq. 4 must be multiplied by the ratio (D/D_*). The static pressure change due to the passage of the expansion wave generated by the opening of the diaphragms in the charge tubes is given by Eq. 2 above. Knowing the ratio of the stagnation pressure to the static pressure, the relationship between the stagnation pressure and the initial charge pressure is given by Eq. 5 and shown graphically in Fig. 18.

$$\frac{p_2^o}{p_r} = \left[\frac{1+\frac{\gamma-1}{2}M_2^2}{\left(1+\frac{\gamma-1}{2}M_2\right)^2}\right]^{\gamma/\gamma-1} \tag{5}$$

since $p_1 = p_r$.

The sharp edged orifices are not 100% efficient based on the actual orifice area (α_o), and a correction factor or orifice coefficient (C_i) must be applied.

$$C_i\alpha_o = \alpha_e \qquad \text{or} \qquad d_e^2 = C_i\, d_o^2$$

The orifice coefficients vary with Reynolds number up to a certain minimum; above this they are independent of Reynolds number and hence remain constant. This critical Reynolds number is shown in Fig. 19 along with a plot of incompressible orifice coefficient versus d_o/d_t above the critical Reynolds number.

The above orifice coefficients are for incompressible flow only, but any flow where $r < 1$ is not incompressible. Jobson[7] has derived compressibility corrected orifice coefficients for subcritical and supercritical flows; Jobson's equations assume a knowledge of the incompressible coefficients:

$$C_D = \frac{1}{2fr^{1/\gamma}}\left[1-\sqrt{1-\frac{(2r^{1/\gamma})^2(1-r)f}{K_N^2}}\right] \tag{6}$$

$$C_D = \frac{1}{2fr_c^{1/\gamma}}\left[\left(1+\frac{(r_c-r)\,r_c^{1/\gamma}}{K_N^2}\right)-\sqrt{\left(1+\frac{(r_c-r)r_c^{1/\gamma}}{K_N^2}-\frac{(2r_c^{1/\gamma})^2(1-r)f}{K_N^2}\right)}\right] \tag{7}$$

These coefficients are presented in Fig. 20. Eq. 6 is for subcritical discharge and Eq. 7 is for supercritical discharge, and

$$f = 1/C_i - 1/2C_i^2$$

An additional phenomena is noticed from Eq. 7, which shows that the supercritical mass flow is dependent on the orifice pressure ratio or downstream pressure. The new effect can be attributed to the action of the orifice pressure ratio on the vena contracta. This differs from the incompressible case where the mass flow is independent of the downstream pressure. Jobson's analytic work has been verified experimentally by Rudinger[8] for the subcritical region.

Unfortunately, the above equations and graphs result in a non-linear combination, and the approximate orifice coefficient must be determined by an iterative procedure.

C. Combustion Chamber

The general design principles for conventional rocket engine combustion chambers are directly applicable to the short-duration combustor. The primary requirement here is that the chamber volume must be large enough that the residence time of the gas mixture allows the reactions to reach completion. In a conventional liquid engine, the fuel and oxidizer enter the combustion chamber in the liquid phase following which they are atomized and mixed. An ignition delay is then experienced during which the mixture absorbs energy to permit vaporization and initiate combustion. In the present combustor, the residence time need not include time for the phase change; hence, criteria suitable for conventional combustors will be conservative with respect to providing sufficient time for the combustion reactions to be complete. A volume or length excessively longer than the minimum leads to cooling penalties which will reduce the overall engine performance and should be avoided, however. The residence time in a chamber of volume V_c is:

$$\tau_c = \frac{V_c}{\dot{m}/\rho_{av}}$$

The equation for the residence time may be stated in terms of the characteristic velocity (c^*) and characteristic length (L^*) of conventional rocket practice.

SECTION IV (Cont'd.) COMBUSTOR DESIGN METHODS

$$c^* = \frac{p_c A_{th}}{\dot{m}} \; ; \; L^* = \frac{V_c}{A_{th}}$$

The theoretical c^* (ideal motor theory) is

$$c^* = \left(\frac{R T_c}{m}\right)^{1/2} \gamma^{-1/2} \left[\frac{\gamma+1}{2}\right]^{\frac{\gamma+1}{2(\gamma-1)}}$$

From the equation of state:

$$\rho_{min} = \frac{p_c m}{R T_c}$$

These equations may be combined to give for the residence time

$$\tau_c = \left(\frac{\rho_{av}}{\rho_{min}}\right) \frac{L^*}{c^*} \frac{1}{\gamma} \left(\frac{\gamma+1}{2}\right)^{\frac{\gamma+1}{\gamma-1}}$$

For the hydrogen-oxygen motor $\gamma \cong 1.15$ and assuming as was done in Ref. 9

$$\frac{\rho_{av}}{\rho_{min}} \cong 2$$

$$\tau_c \cong 4.9 \frac{L^*}{c^*}$$

The prototype combustor has an internal volume of about 125 in^3 and feeds six nozzles with a throat diameter of 0.598 inches. L^* is then calculated to be 74 inches. Summerfield[9] suggests that the minimum value of L for a liquid oxygen, liquid hydrogen engine operating at 300 psi is about 10 inches and that it corresponds to the fundamental reaction time. Consequently, there appears to be adequate margin on the prototype combustor. It also follows that, assuming the reaction kinetics are of second order, the time to completion would vary inversely as the pressure. Accordingly, L^*_{min} must be increased if the combustor is operated at lower pressure. On this basis, the present combustor could then be operated down to about 50 psi chamber pressure and still provide sufficient residence time for the reactions to be completed.

Using values of L^* of 74 inches and c^* of 7500 ft/sec, a theoretical residence time of 5.0 milliseconds is calculated. It should be noted that for the constant pressure combustor being used, the residence time is identical to the time from the moment flow commences into the combustion chamber to the time the design pressure and nozzle diaphragm rupture takes place assuming that combustion is initiated as soon as inflow commences. From Fig. 9 it may be seen that this time is actually about 3 milliseconds. The difference in measured and calculated values would

SECTION IV (Cont'd.) COMBUSTOR DESIGN METHODS

indicate that the value of 2 for $\rho_{av.}/\rho_{min.}$, while a good approximation for a liquid injected engine is too large for the present case. This would imply that most of the gases in the combustion chamber are in the reacted state. In view of the relatively large value of L^*, such would be expected to be the case.

A knowledge of the combustion efficiency is necessary for the accurate interpretation and application of the experimental data taken in the base heating program. In principle, the combustion efficiency can be calculated from a knowledge of the combustor geometry and the measured charge tube and combustion chamber pressure. Assuming that steady state conditions have been reached, the following mass flow balance equation can be written:

$$\dot{m}_{H_2 \text{ orifice}} + \dot{m}_{O_2 \text{ orifice}} = \dot{m}_{\text{nozzles}}$$

The method of calculating the mass flow through the charge tube orifices was developed in a previous section. The mass flow through the choked nozzles may be calculated from the following:

$$\dot{m} = \left(\frac{\gamma \mathcal{M}_{cc}}{g R T^{o}_{cc}}\right)^{1/2} D_* \alpha_{NT} n_N P^{o}_{cc}$$

The measured combustion chamber pressure which is a static pressure can be used instead of the stagnation pressure. The static and stagnation pressures differ by 1.5% for the combustor being considered. The Mach number function D_* has the following values:

γ	D_*
1.1	.5992
1.2	.5921
1.3	.5851
1.4	.5787

The combustion chamber temperature can be found from the following equation:

$$T^{o}_{cc} = \frac{\gamma_{cc}}{\gamma_{H_2}\, \varepsilon'_{O_2}} \frac{(D_*)^2 n_N^{\,2} d_{NT}^{\,4} P^{o\,2}_{cc} (\mathcal{M})_{cc} T_o}{\left[(\mathcal{M})^{1/2}_{O_2} (d_e)^2_{O_2} D_{o_{O_2}} + (\mathcal{M})^{1/2}_{H_2} (d_e)^2_{H_2} D_{o_{H_2}}\right]^2}$$

All parameters in the above equation can be computed from previous sections except γ_{cc} and m_{cc}. These parameters are dependent on the temperature and pressure in the combustion chamber. It is possible to use their theoretical values if the efficiency is near 100%. Refs. 10 and 11 may be used to determine equilibrium composition of a hydrogen-oxygen system. If the efficiency is not near 100%, an iterative process will be necessary to determine T°_{cc}. The pressures being considered are relatively high and the residence times long; it is assumed in this report that the flow is in equilibrium.

The efficiency of combustion is defined as the ratio of the actual energy released to the ideal or maximum enthalpy released by the fuel:

$$\eta = \frac{\Delta h_{cc}}{\Delta h_{th}} = \frac{h_{cc} - h_o}{h_{th} - h_o}$$

The ideal enthalpy of a stoichiometric mixture of hydrogen and oxygen is 6927.0 BTU/lb. Ideal enthalpy for other 0/F ratios can be computed. Enthalpy for equilibrium composition at combustion temperature and pressure can be found in Ref. 10.

In practice, this is not an acceptable means of determining combustion efficiency since a small error in pressure measurement leads to very large errors in the calculation of the other quantities. An error of 10% in the ratio p_{cc}/p_c, for example, will result in errors of approximately 50%, 25% and 36% in combustion efficiency, temperature and enthalpy, respectively. Consequently, these calculation procedures should be used only for combustor design purposes or very gross calculations. These methods were used to verify a final combustor design with the results shown in Fig. 21. The experimental relations between charge tube pressure and combustion chamber pressure are seen to agree to reasonable accuracy.

Various unstable combustion phenomena are observed in liquid propellant rocket motors which are discussed in standard references.[9,12] No stability analyses have been performed in connection with the current development since at no time have any instabilities at either high or low frequency been observed. The piezoelectric transducers and electronic recording system used for combustion chamber pressure measurement have excellent frequency response being flat to at least 5 KC. The good response characteristics are evident from Figs. 8 and 9. Inspection of Fig. 9 shows that the combustion is very steady.

D. Mixer Plate - Injector

The mixer plate has the functions of introducing the fuel and oxidizer into the combustion chamber, of mixing the two as completely as possible, and distributing the gases in the combustion chamber. Since the propellants are already in a gaseous state, there is no need for atomization so that a very simple, direct impingement-of-streams approach is employed with many pairs of impinging fuel and oxidant gas streams.

The mass flow into the combustion chamber is conveniently controlled by choking the flow at some point in the system. This metering could be done in the mixing plate. In this particular case, however, it appeared to be satisfactory to do the metering with choked orifices at the downstream end of the charge tubes and maintain subcritical flow through the mixing plate. This had the advantages of reducing the overall pressure drop, permitting changes in 0/F ratio simply by changing orifices in the charge tubes and minimizing the flow calibrations required. One possible advantage of choking in the mixer plate, however, might be to insure an equal mass flow through all of the orifices. It appears that either method is probably satisfactory.

E. Cutter Assembly

The cutter assembly for rupturing the supply tube diaphragms is generally operated by an impact device. That is, a spring-loaded mass is accelerated and impacts on a cam which, in turn, translates or rotates a shaft on which are mounted the cutter blades. Three important considerations for this assembly are: (1) it must operate very rapidly (preferably less than 1/2 millisecond); (2) the shaft should be adequately sealed between the atmosphere and the supply tubes; (3) the cutter blades should have adequate travel before touching the diaphragms so that they do not cut until the mechanism is triggered; (4) the cutter blades should have enough energy to cut through the diaphragms.

SECTION V CONCLUSIONS AND RECOMMENDATIONS

A. Short-duration combustors have been developed which may be used to simulate liquid propellant rocket engine exhaust gases.

B. Either constant volume or constant pressure devices may be used with generally satisfactory results.

C. The constant volume combustor requires that the gases be thoroughly mixed in the combustion tube and that many ignition sources be used to prevent detonation waves from developing. A simple plenum chamber downstream of the tube is sufficient to feed the exhaust nozzles. The combustion pressure ruptures the main diaphragm; no mechanical cutters are required. The primary disadvantage is that heat transfer to the combustor walls results in a decrease in total pressure and enthalpy during a run.

D. Either premixed or separate charge gases may be used with the constant pressure combustor. The premixed gas arrangement presents difficulties with flashback into the charge tube and is more hazardous to use. Its only advantage appears to be a slightly shorter overall length for the same test time.

E. The optimum combustor developed to date is a constant pressure device using separate charge tubes for the fuel and oxidizer gases. Diaphragms are placed in the exit nozzles which do not burst until the design pressure has been reached. Ignition is accomplished by the adiabatic temperature rise of the combustible mixture flowing into the combustion chamber upon rupture of the charge tube diaphragms.

F. No combustion instabilities have been observed with the constant pressure combustor.

G. Measured combustor performance agrees well with the calculated performance.

H. Good parameter control and a wide performance range is possible. 0/F ratio may be changed by changing orifice plates. The combustor operates satisfactorily over a wide range of pressures.

LIST OF SYMBOLS

a	Speed of sound
C_i	Incompressible orifice coefficient
C_D	Compressible orifice coefficient
d_e	Effective orifice diameter
d_{NT}	Nozzle throat diameter
d_o	Orifice diameter
d_t	Supply tube diameter
D	Mach number function = $M\left(1+\frac{\gamma-1}{2}M^2\right)^{-\frac{\gamma+1}{2(\gamma-1)}}$ (see Fig. 14 and Ref. 6)
D_*	Mach number function at M = 1 (= .5787@ γ = 1.4)
D_2	Mach number function in supply tube near orifice after passage of expansion wave
h_{cc}	Enthalpy in combustion chamber
h_o	Enthalpy of supply gas
h_{th}	Theoretical enthalpy
K_N	$= \sqrt{\gamma\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}$
ℓ	Length of supply tube
M	Mach number
$\dot{m}$	Mass flow rate
$\mathcal{M}$	Molecular weight
n_N	Number of nozzles
η	Efficiency of combustion
p	Static pressure
p_c	Combustion chamber pressure
p_r	Charge tube pressure
p°	Stagnation pressure
r	Pressure ratio across the orifice
r_c	Critical pressure ratio across the orifice = $\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma}{\gamma-1}}$
t	Test time
T	Static temperature
T°	Stagnation temperature
$\dot{w}$	Weight flow rate

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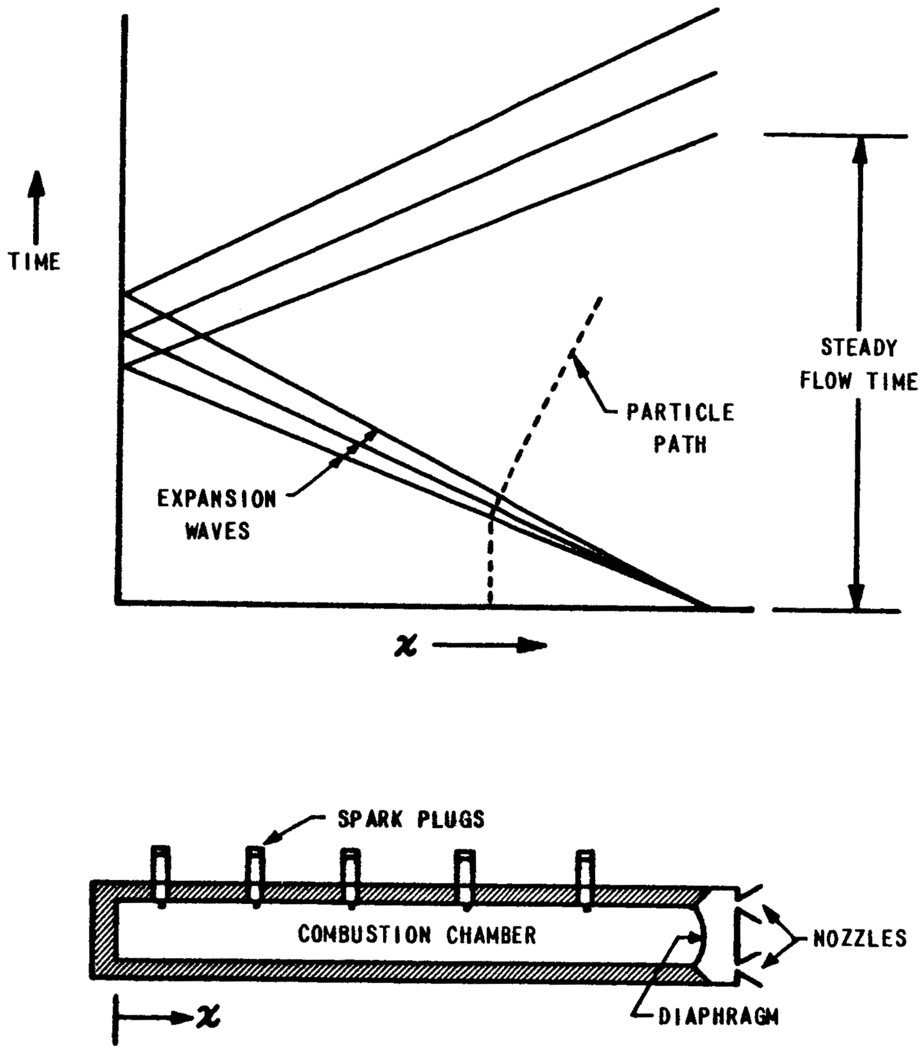
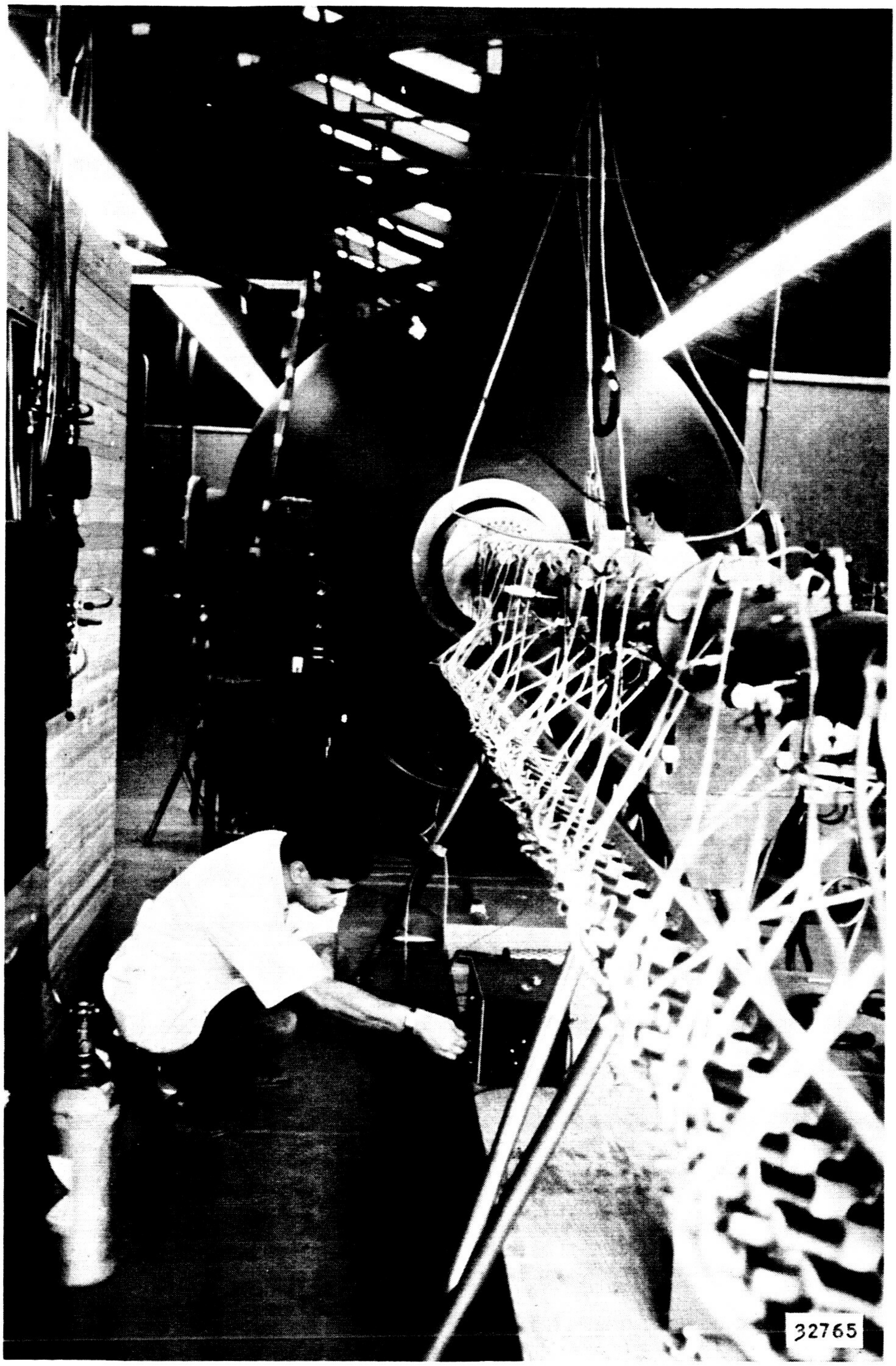


Figure 1 CONSTANT VOLUME COMBUSTOR, SHORT DURATION FLOW GENERATOR



PHOTOGRAPH OF CONSTANT VOLUME COMBUSTOR

Figure 2

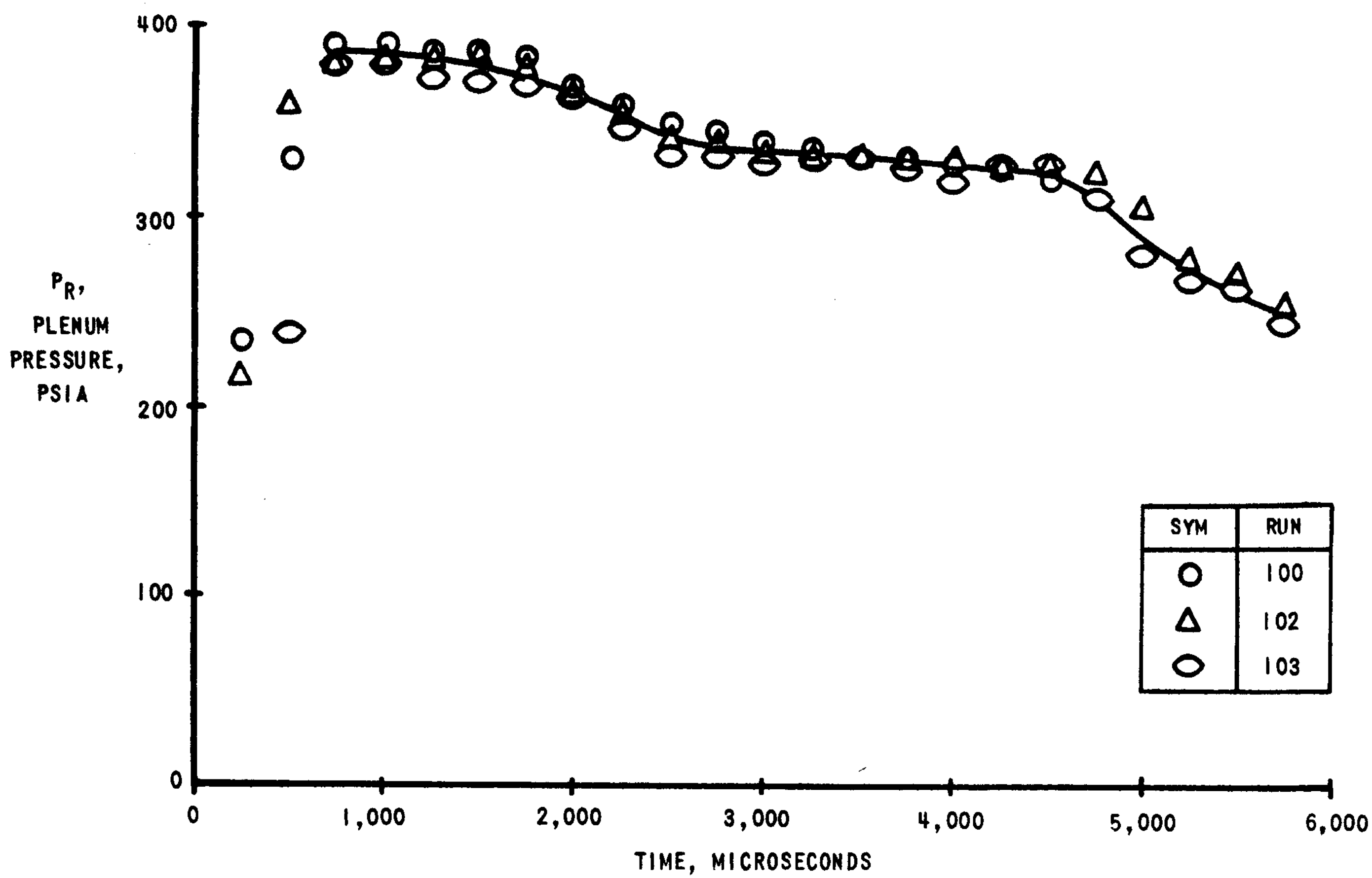
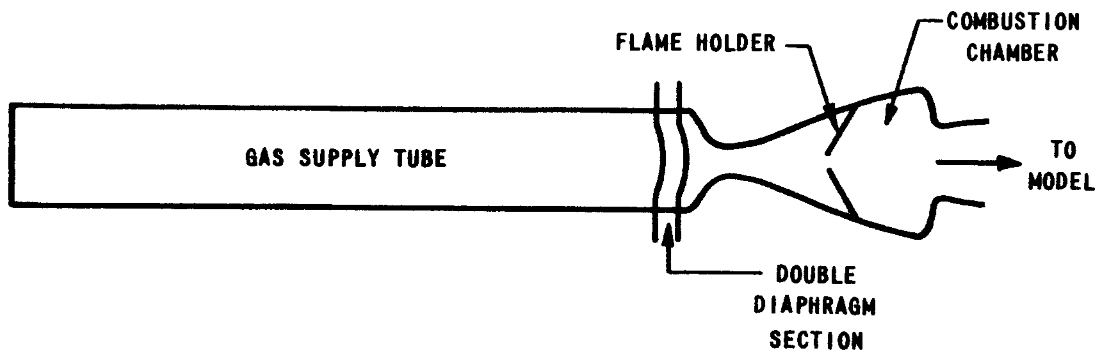
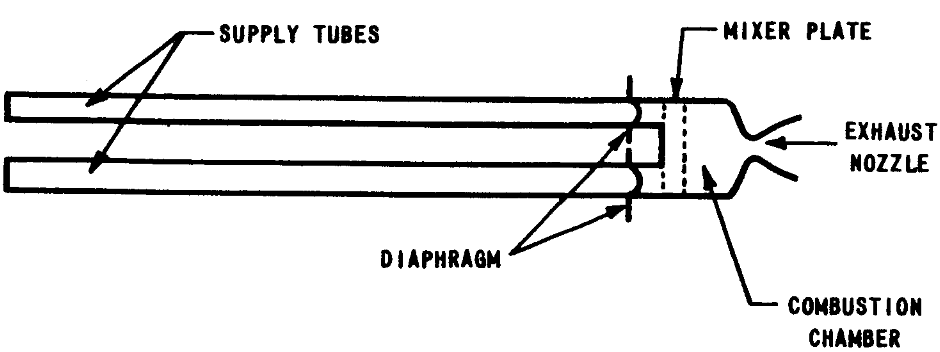


Figure 3 PLENUM PRESSURE VARIATION WITH TIME FOR THREE RUNS USING CONSTANT VOLUME COMBUSTOR



PREMIXED GAS SUPPLY



SEPARATE GAS SUPPLIES

Figure 4 SCHEMATIC ILLUSTRATION OF CONSTANT PRESSURE COMBUSTORS

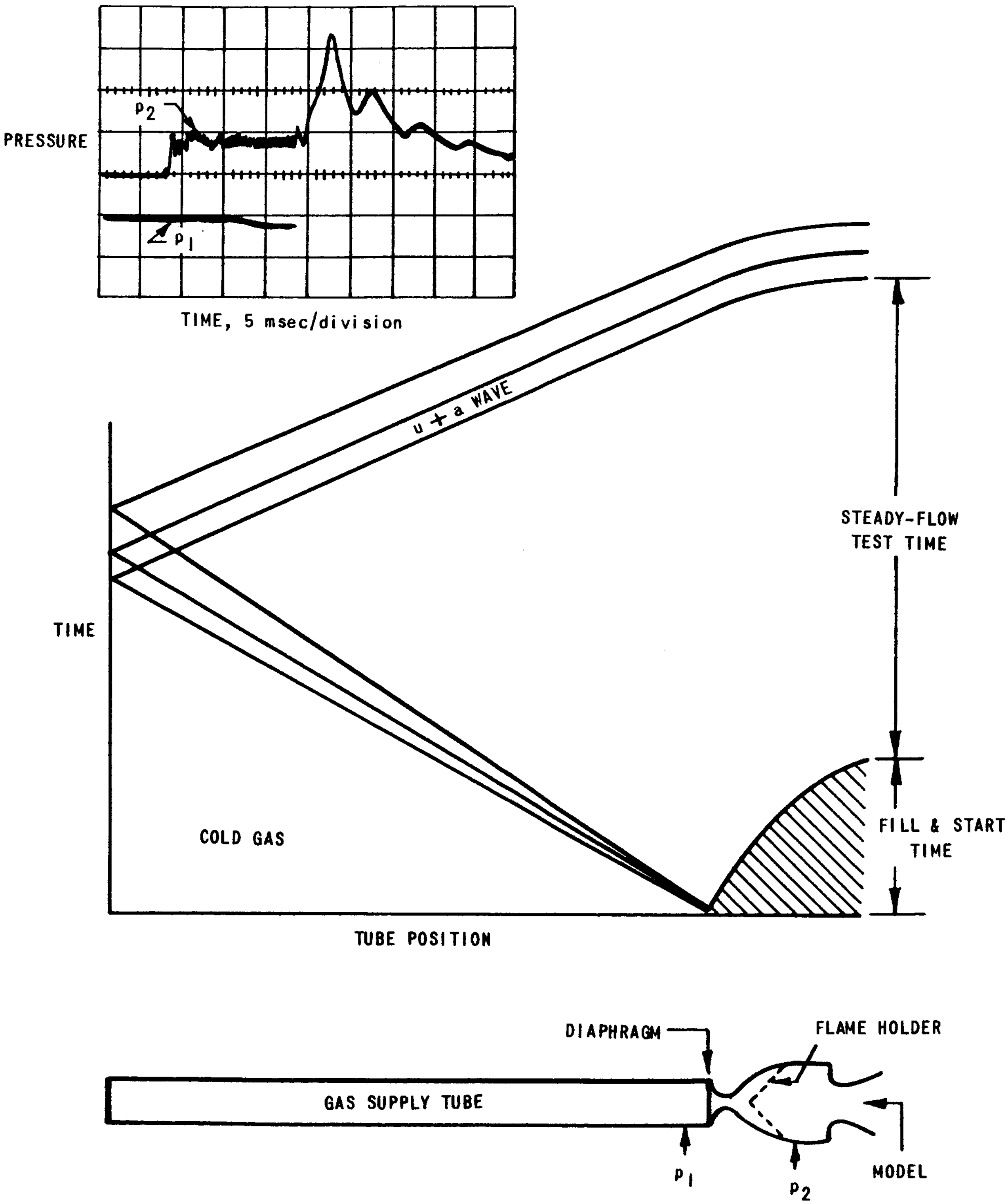
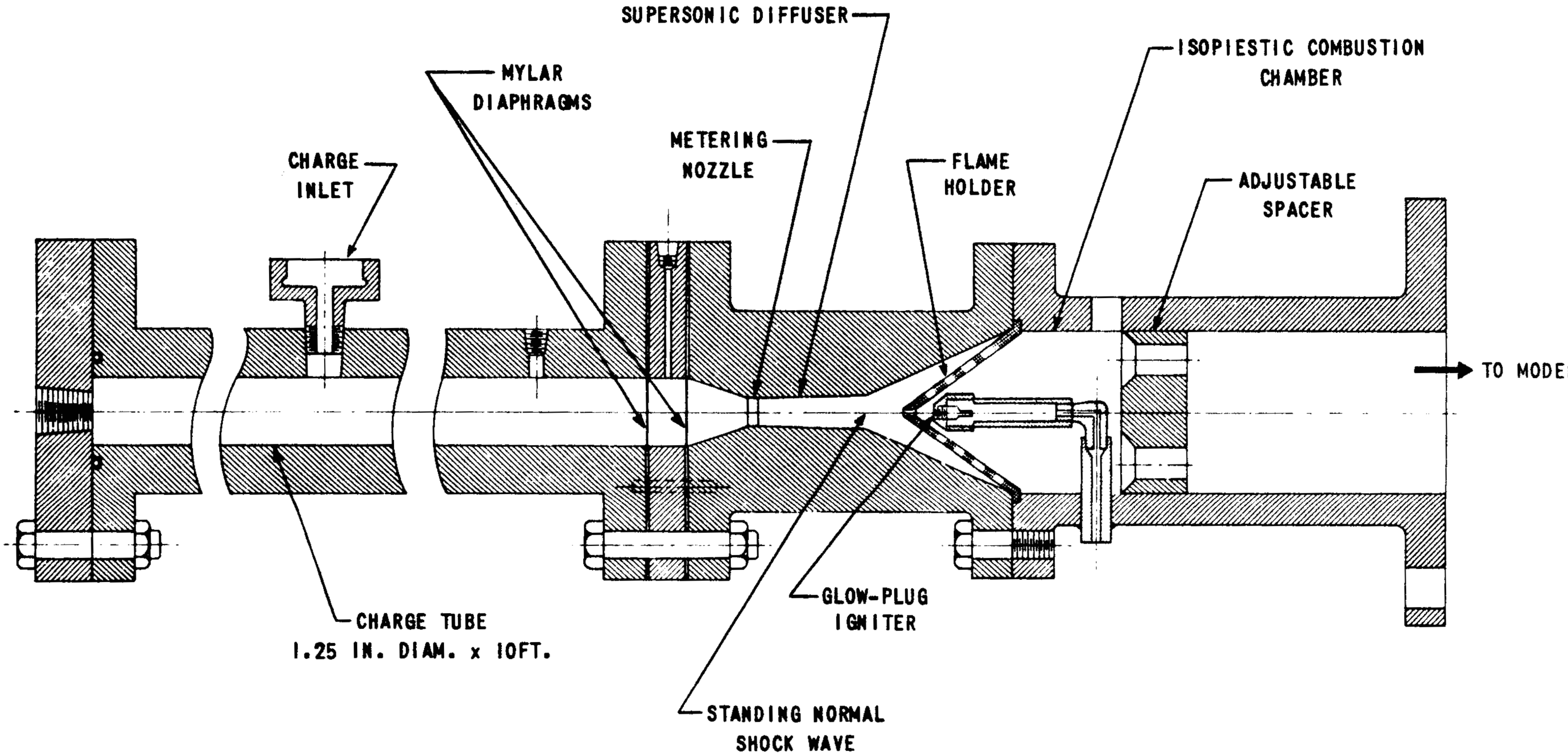
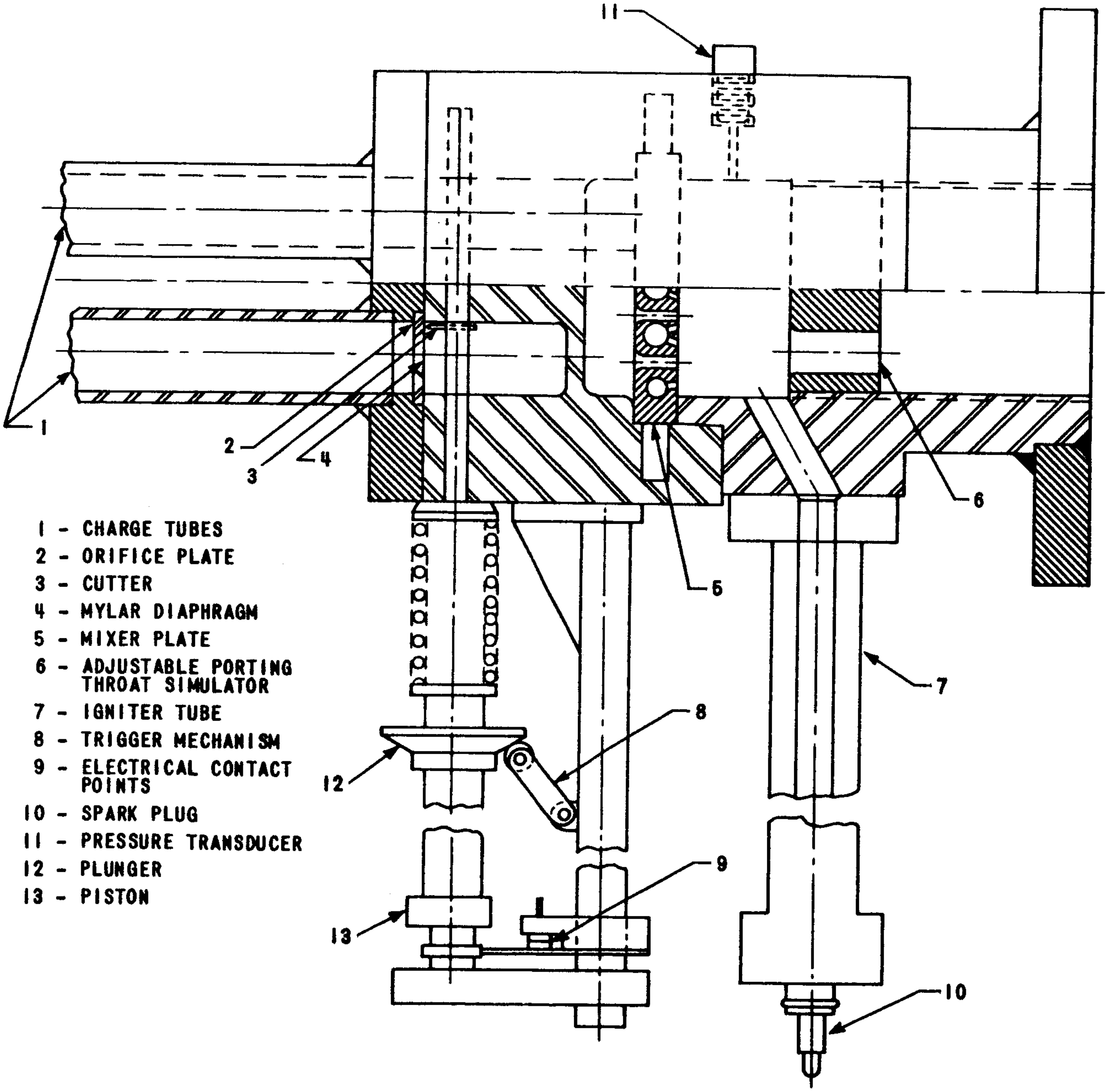


Figure 5 CONSTANT PRESSURE COMBUSTOR, SHORT DURATION FLOW GENERATOR



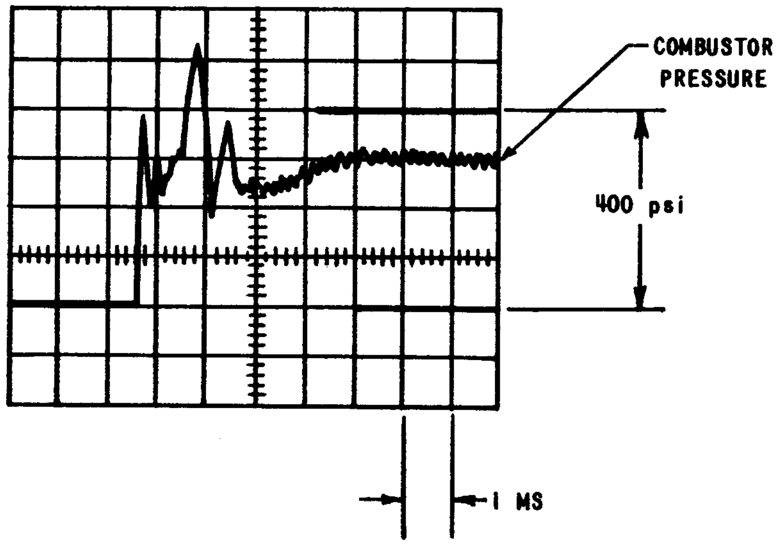
ISOPIESTIC COMBUSTOR FOR SATURN BASE HEATING MODEL

Figure 6



PROTOTYPE CONSTANT PRESSURE COMBUSTOR

Figure 7



CHARGE TUBE PRESSURE - 495 psig
IGNITOR PRESSURE - 430 psig
O_2/H_2 = 4.95 BY WEIGHT

Figure 8 TYPICAL PRESSURE HISTORY FOR PROTOTYPE COMBUSTOR

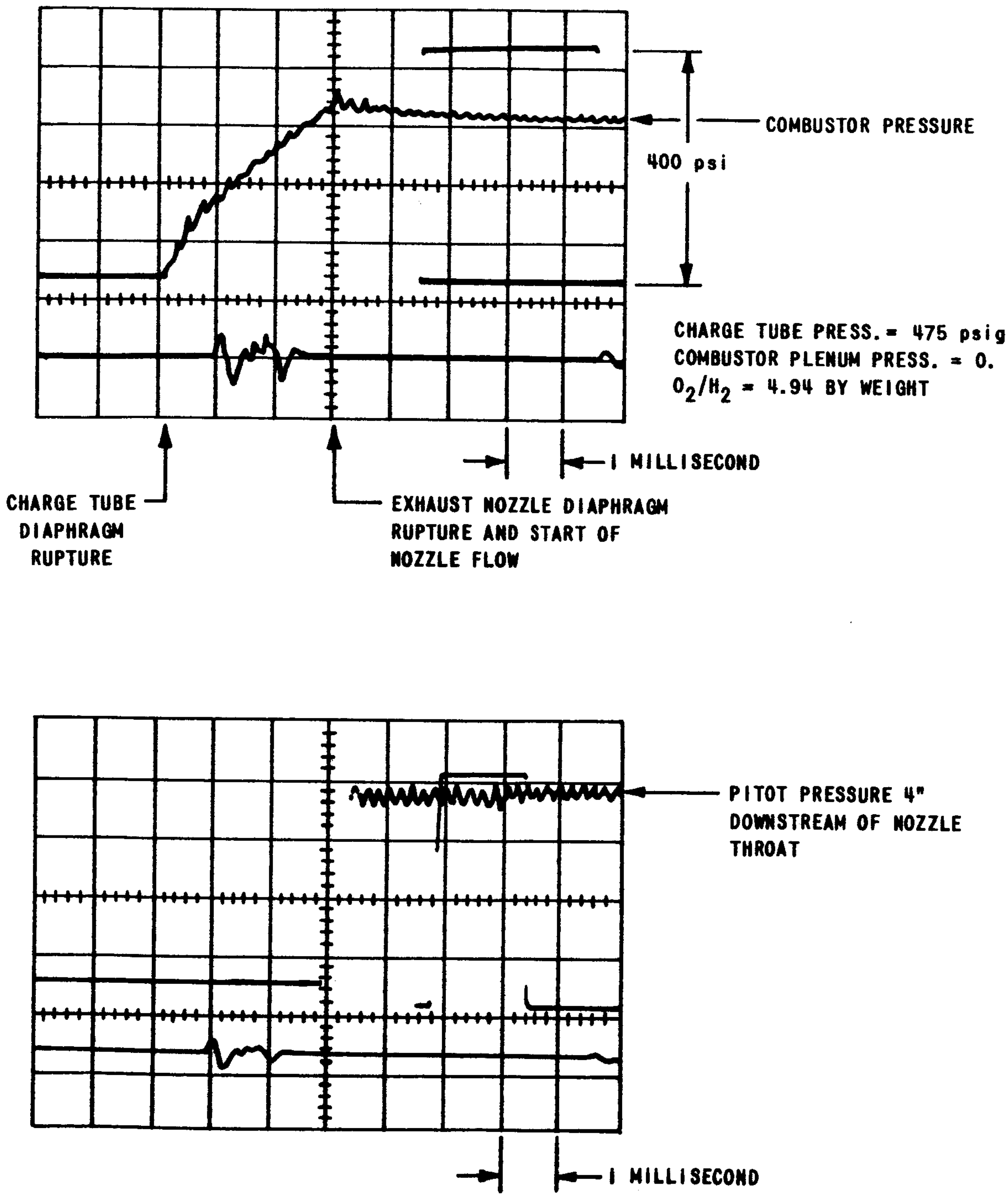
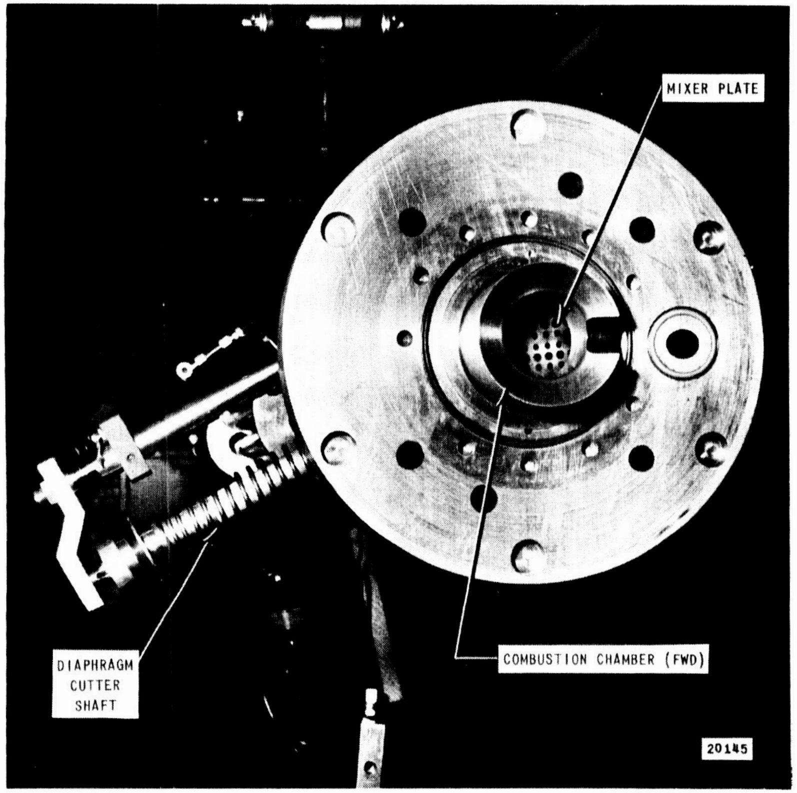
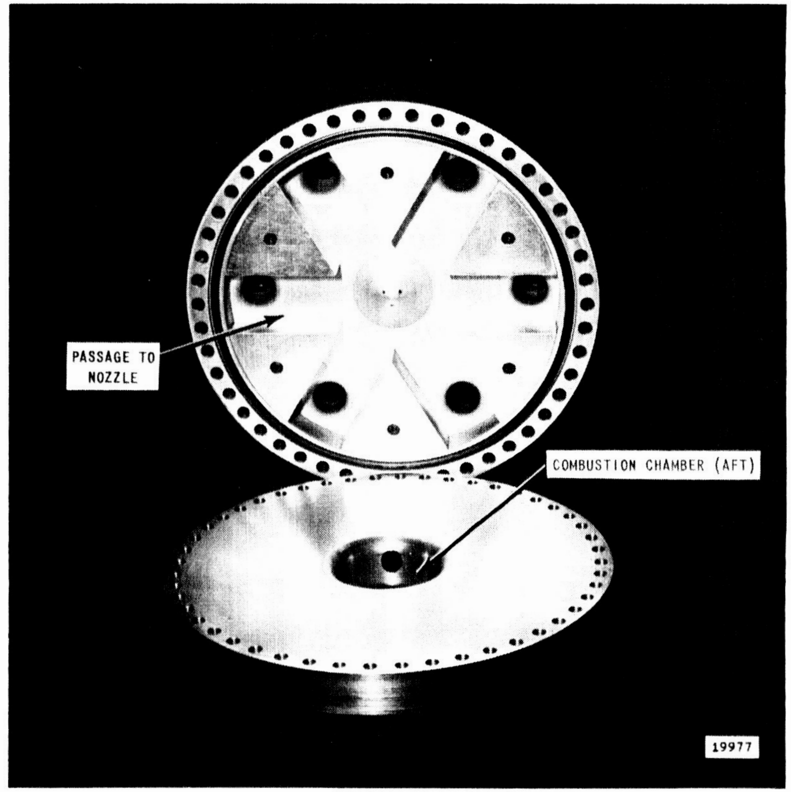


Figure 9 CONSTANT PRESSURE COMBUSTOR OPERATION WITH AUTO-IGNITION

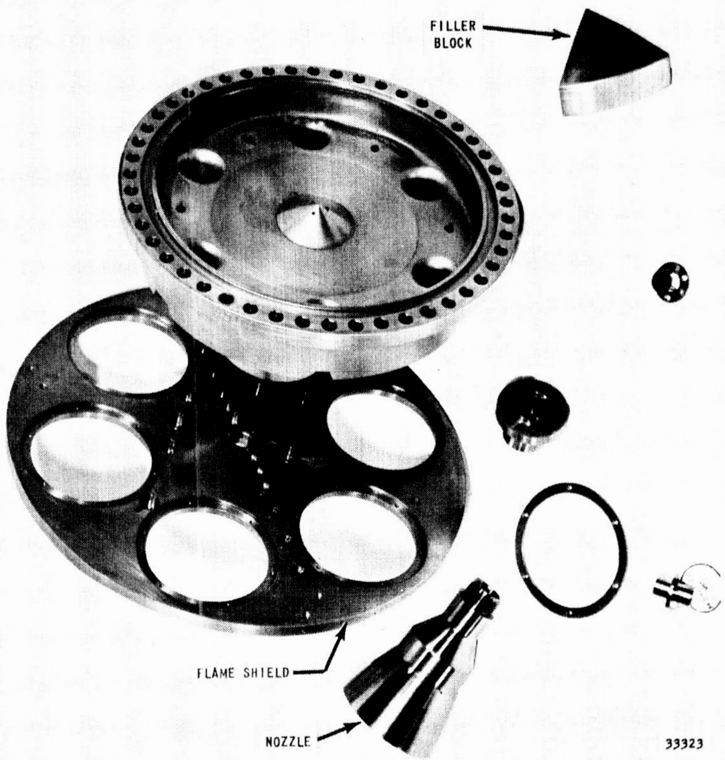


a. FORWARD ASSEMBLY

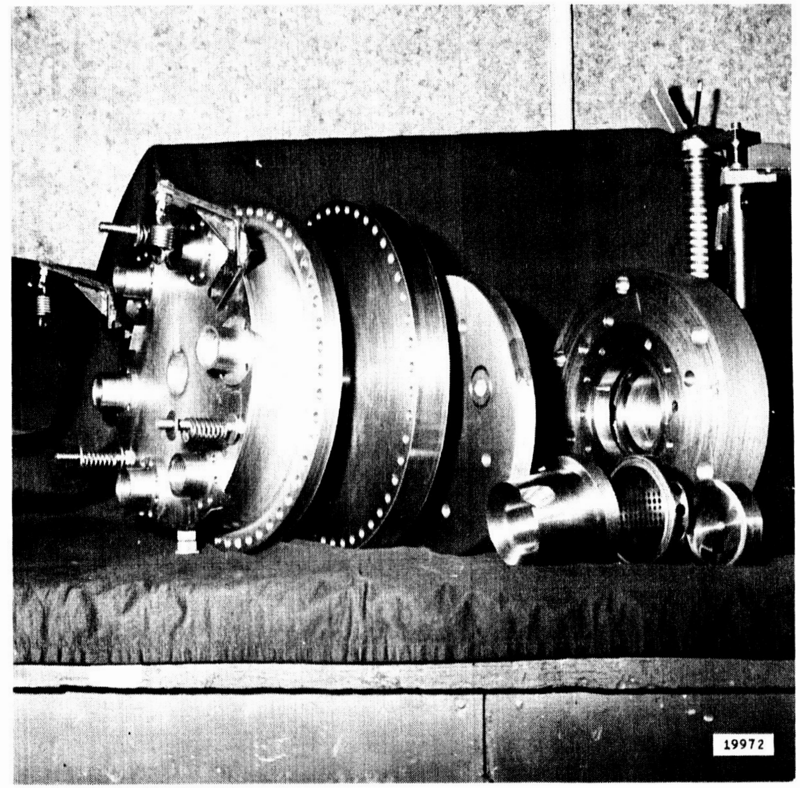


b. AFT COMPONENTS

Figure 10 COMPONENTS OF CONSTANT PRESSURE COMBUSTOR

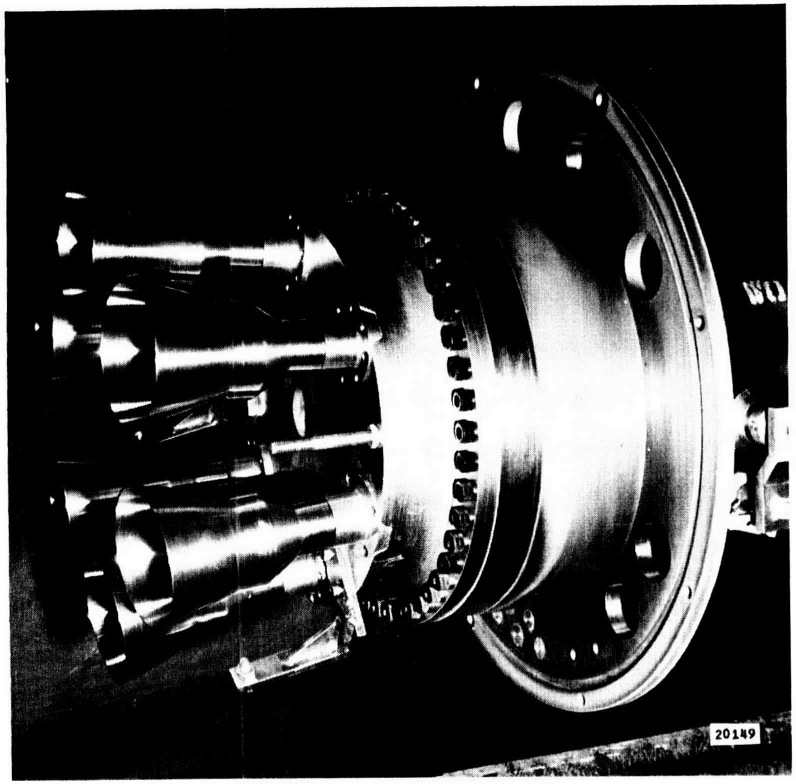


c. AFT COMPONENTS



d. ARRANGEMENT OF PARTS

Figure 10 (Cont.) COMPONENTS OF CONSTANT PRESSURE COMBUSTOR



e. ASSEMBLED COMBUSTOR

f. REAR VIEW

Figure 10 (Concl.) COMPONENTS OF CONSTANT PRESSURE COMBUSTOR

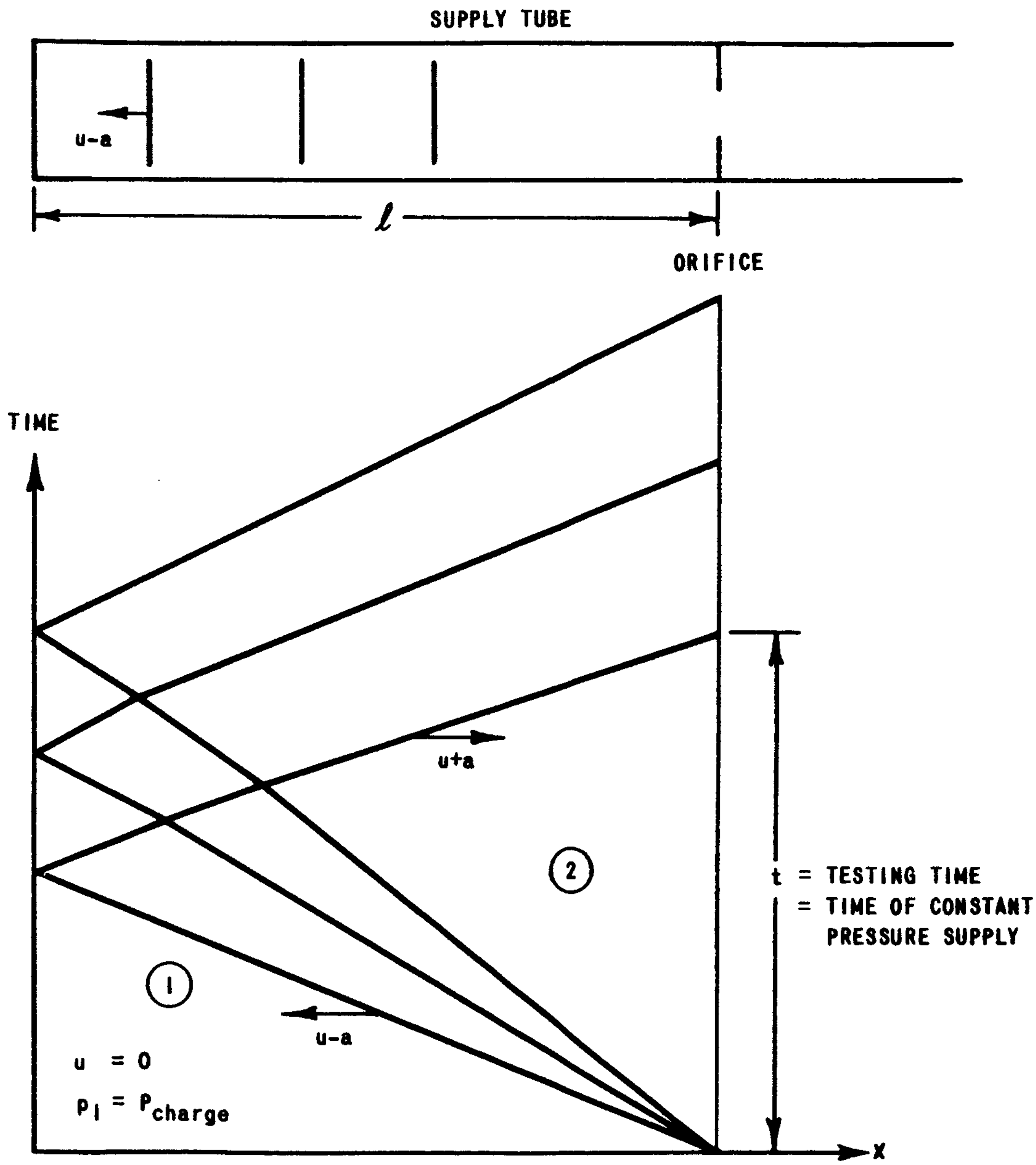


Figure 11 WAVE DIAGRAM OF SUPPLY TUBE

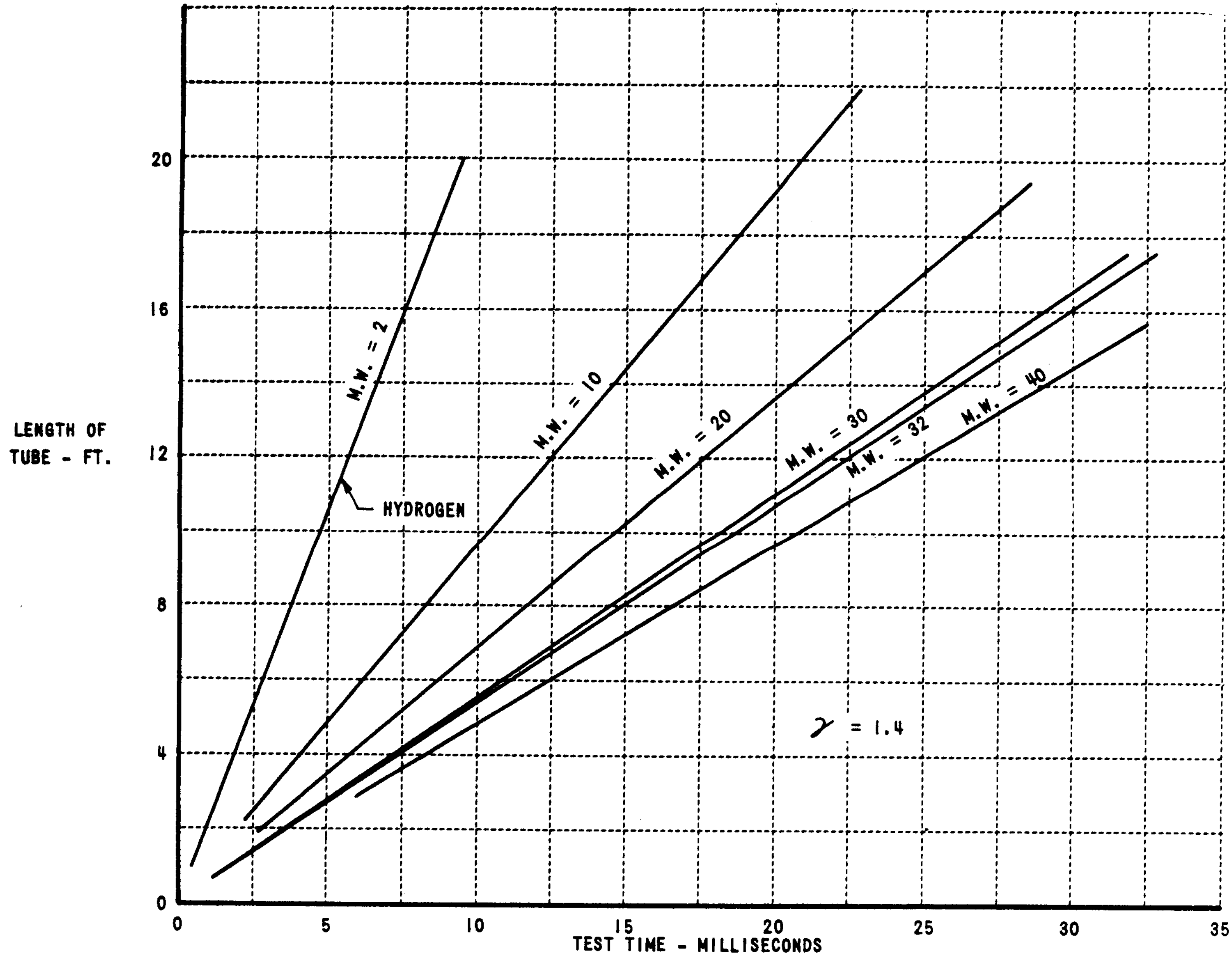


Figure 12 CHARGE TUBE LENGTH FOR VARIOUS MOLECULAR WEIGHT GASES

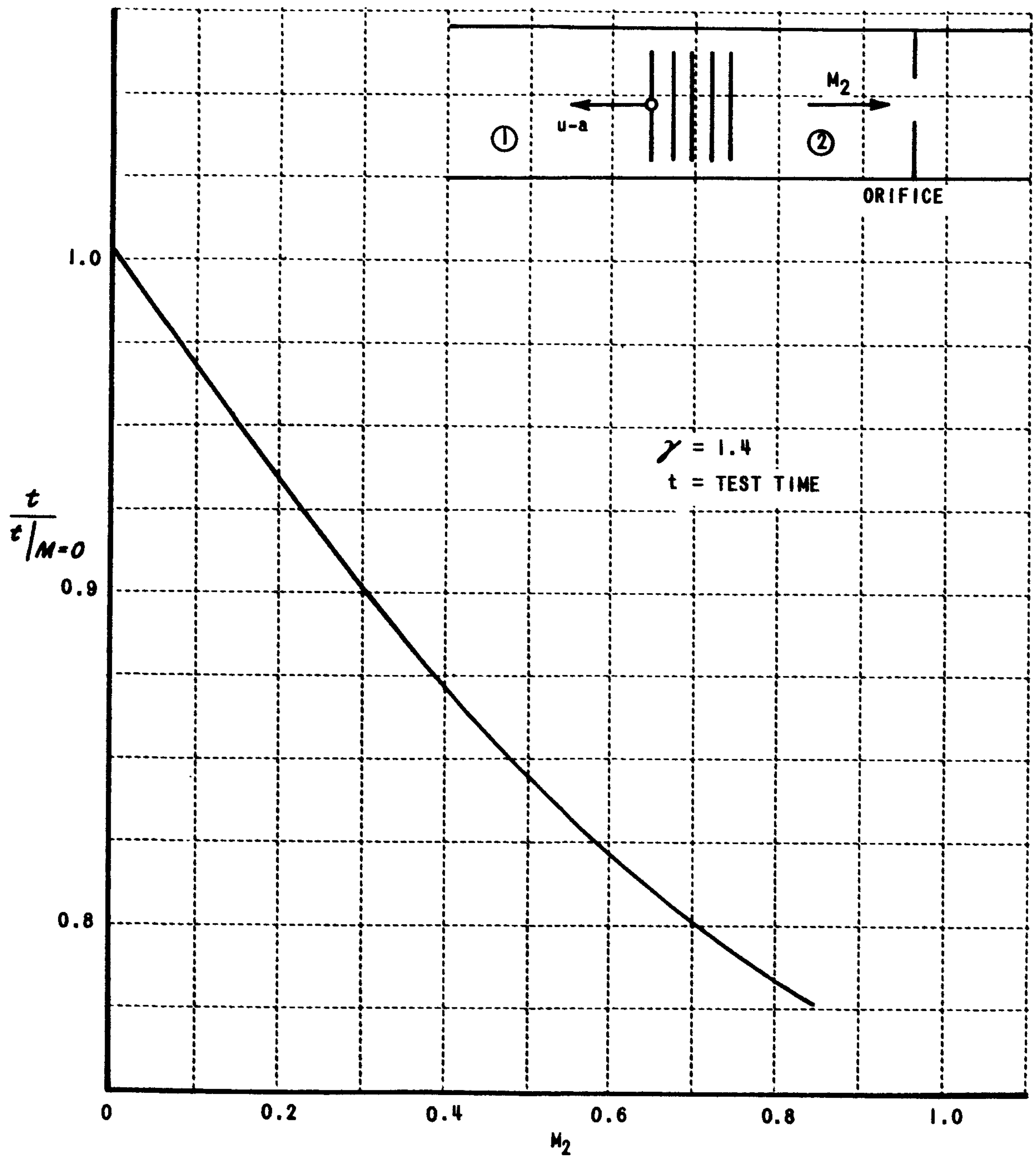


Figure 13 EFFECT OF SUPPLY TUBE MACH NUMBER ON TESTING TIME

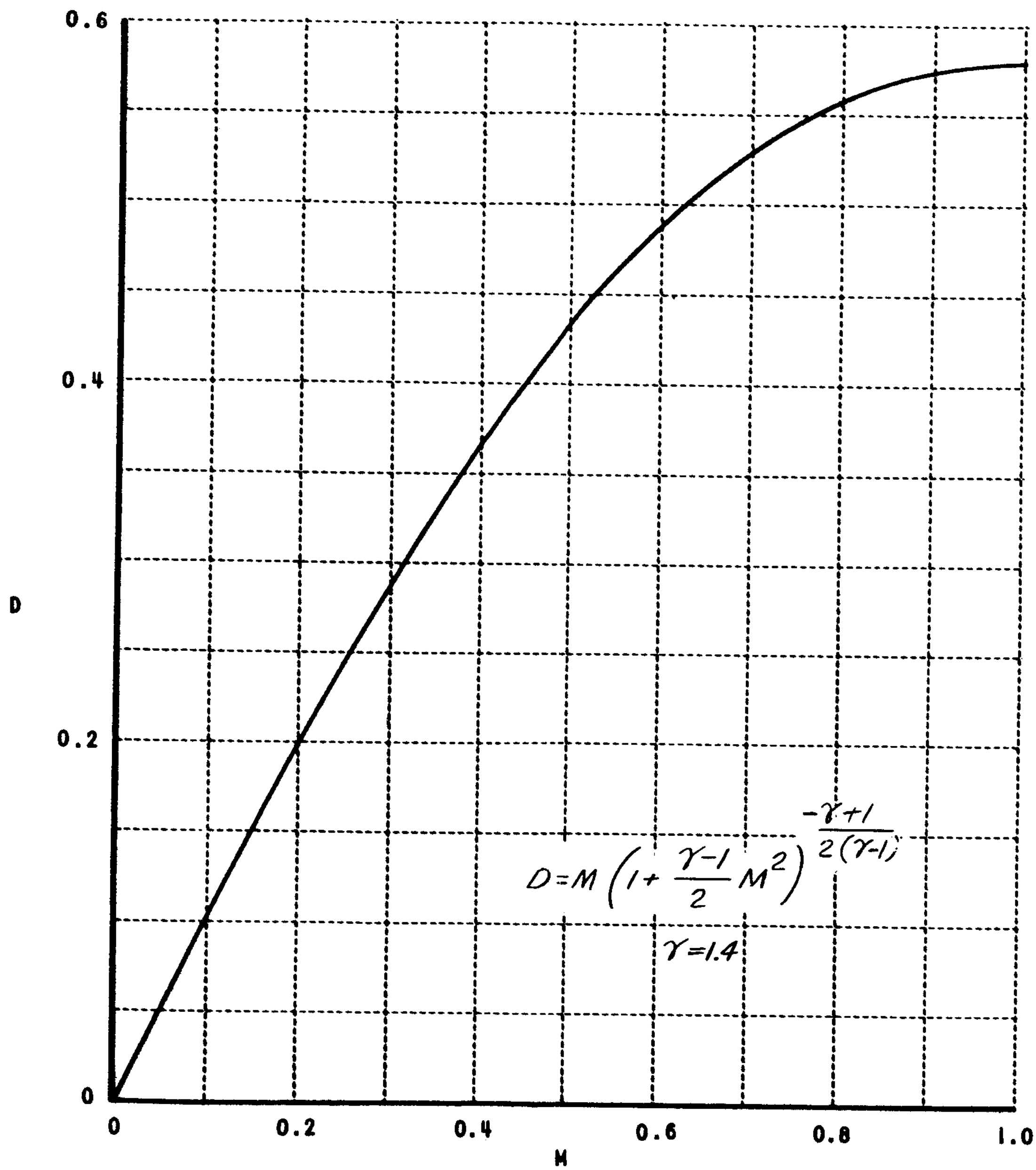


Figure 14 MACH NUMBER FUNCTION D VS MACH NUMBER

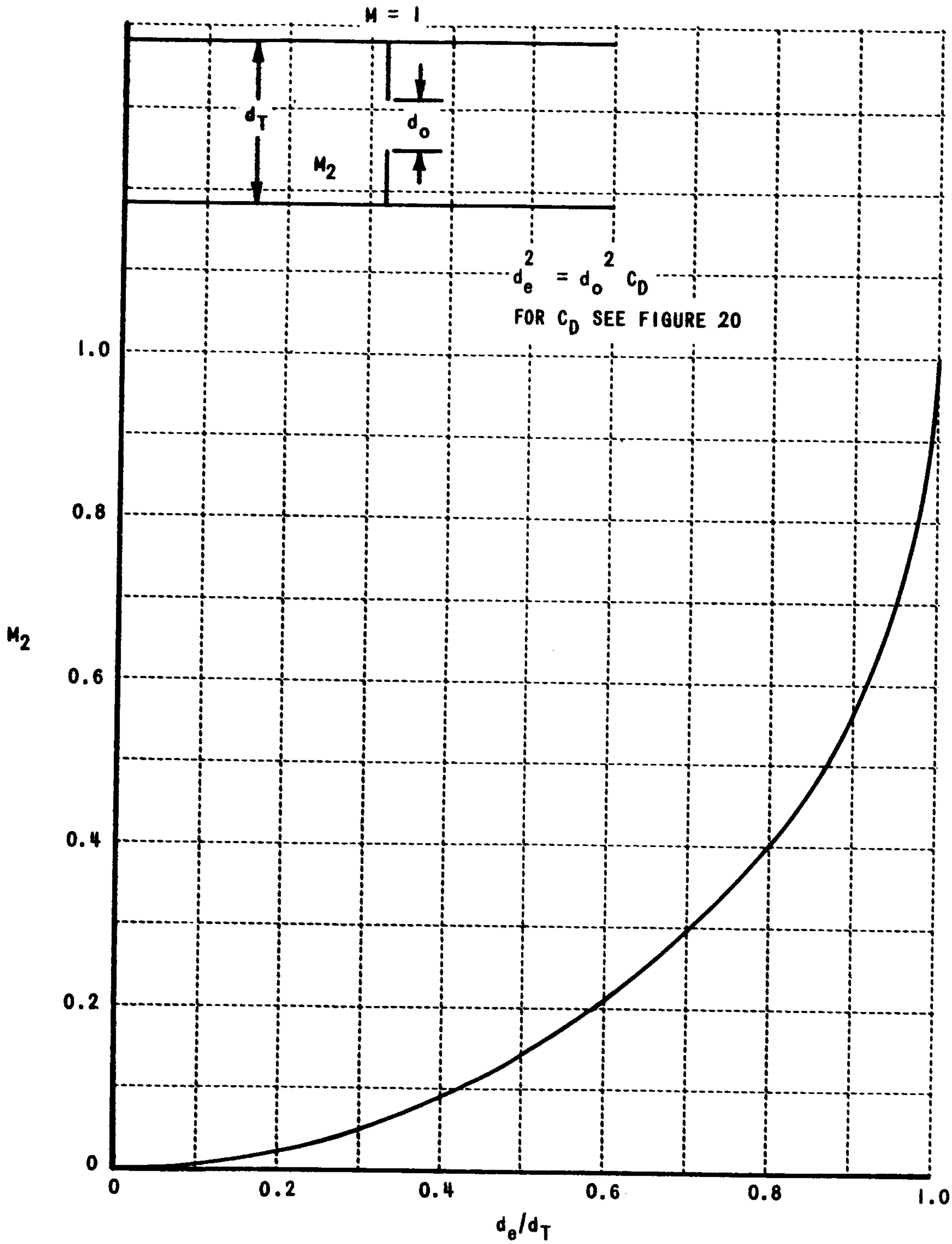


Figure 15 MACH NUMBER IN SUPPLY TUBE VS EFFECTIVE ORIFICE TO TUBE DIAMETER RATIO FOR CHOKED FLOW

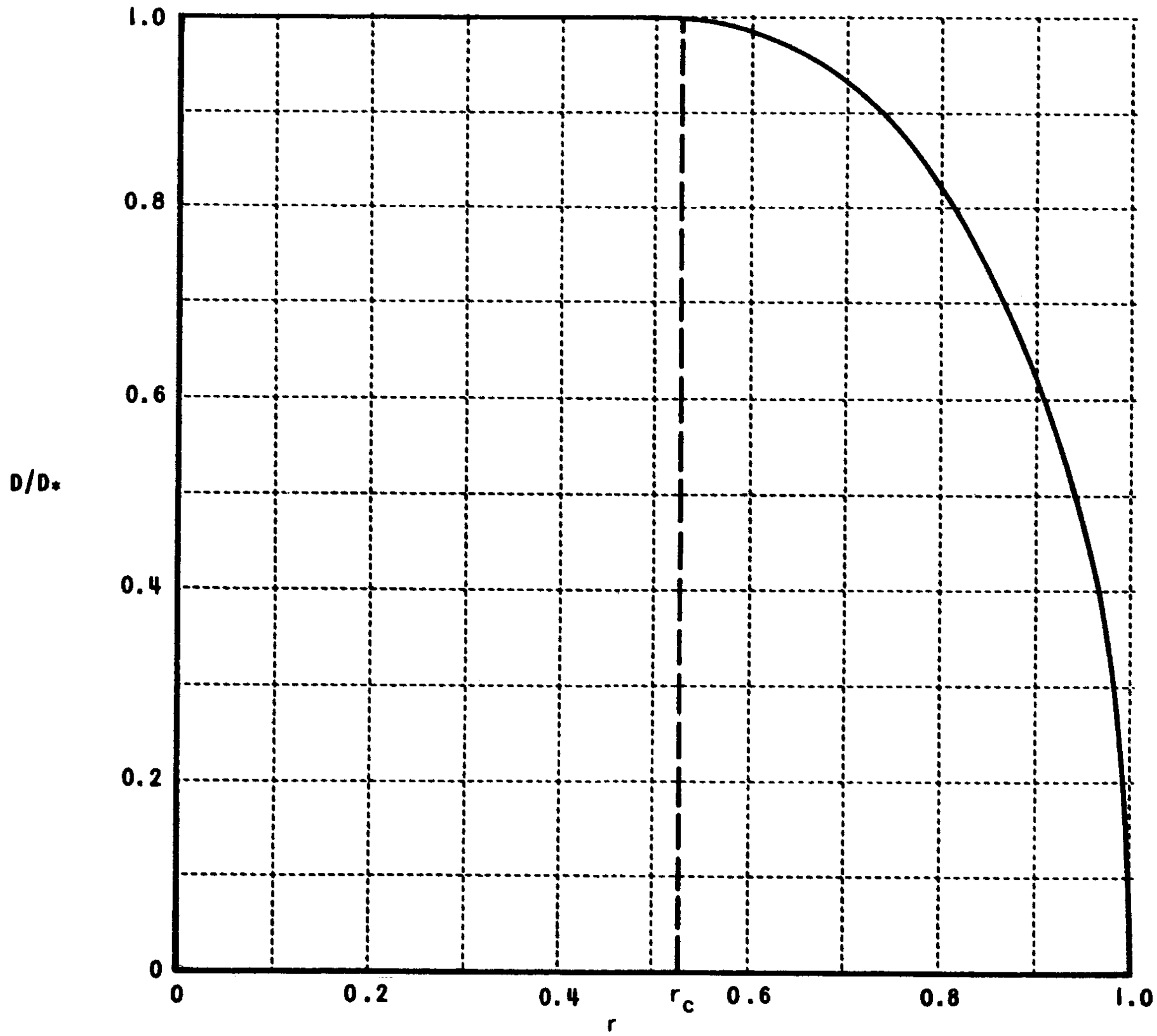


Figure 16 D/D_* VS r

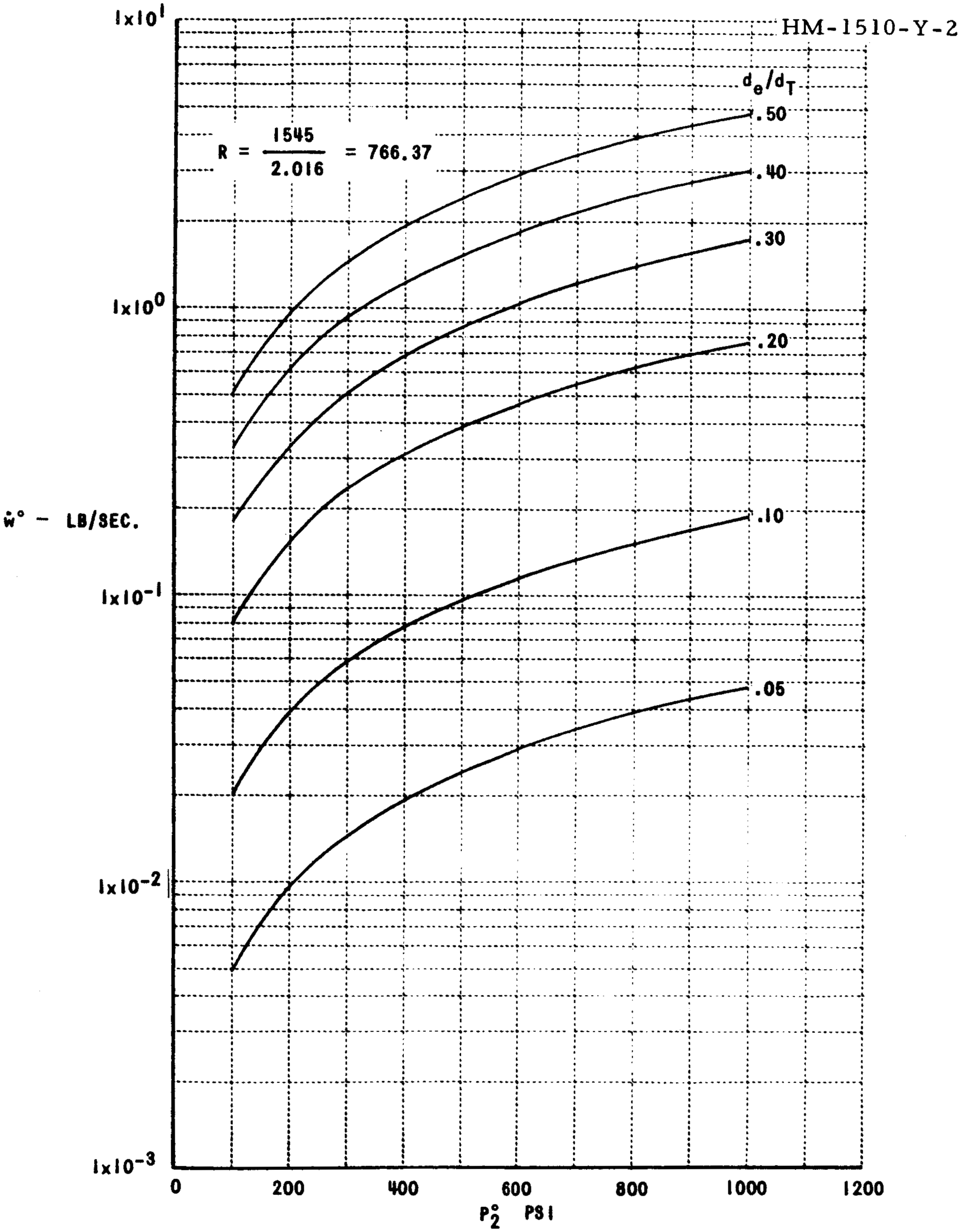


Figure 17a HYDROGEN WEIGHT FLOW THROUGH A CHOKED ORIFICE VS STAGNATION PRESSURE OF SUPPLY AT CONSTANT d_e/d_T RATIOS FOR t^o = 530°R AND d_T = 1 INCH

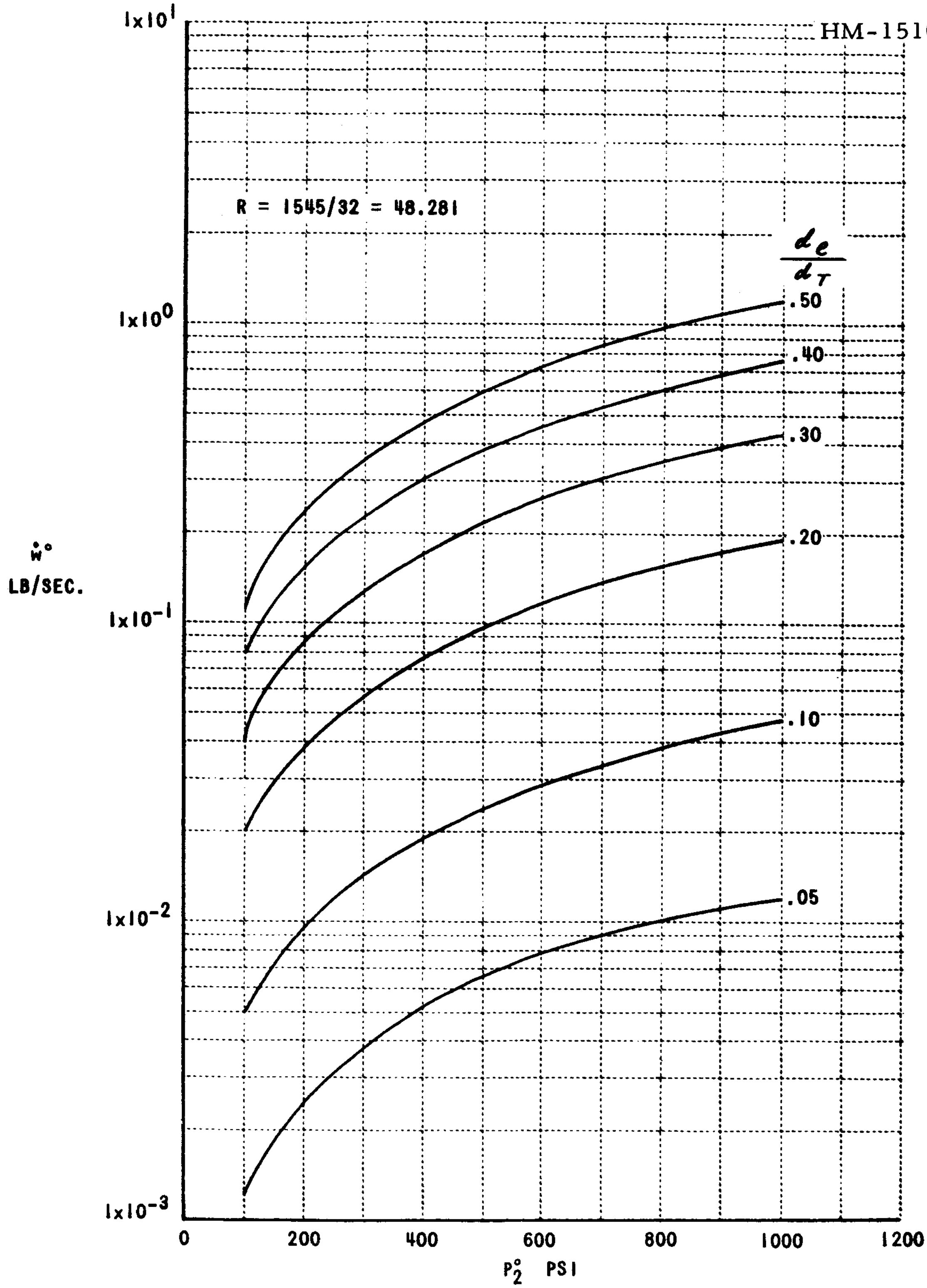


Figure 17b OXYGEN WEIGHT FLOW THROUGH A CHOKED ORIFICE VS STAGNATION PRESSURE OF SUPPLY AT CONSTANT d_e/d_T RATIOS FOR $T^o = 530^oR$ AND d_T = 1 INCH

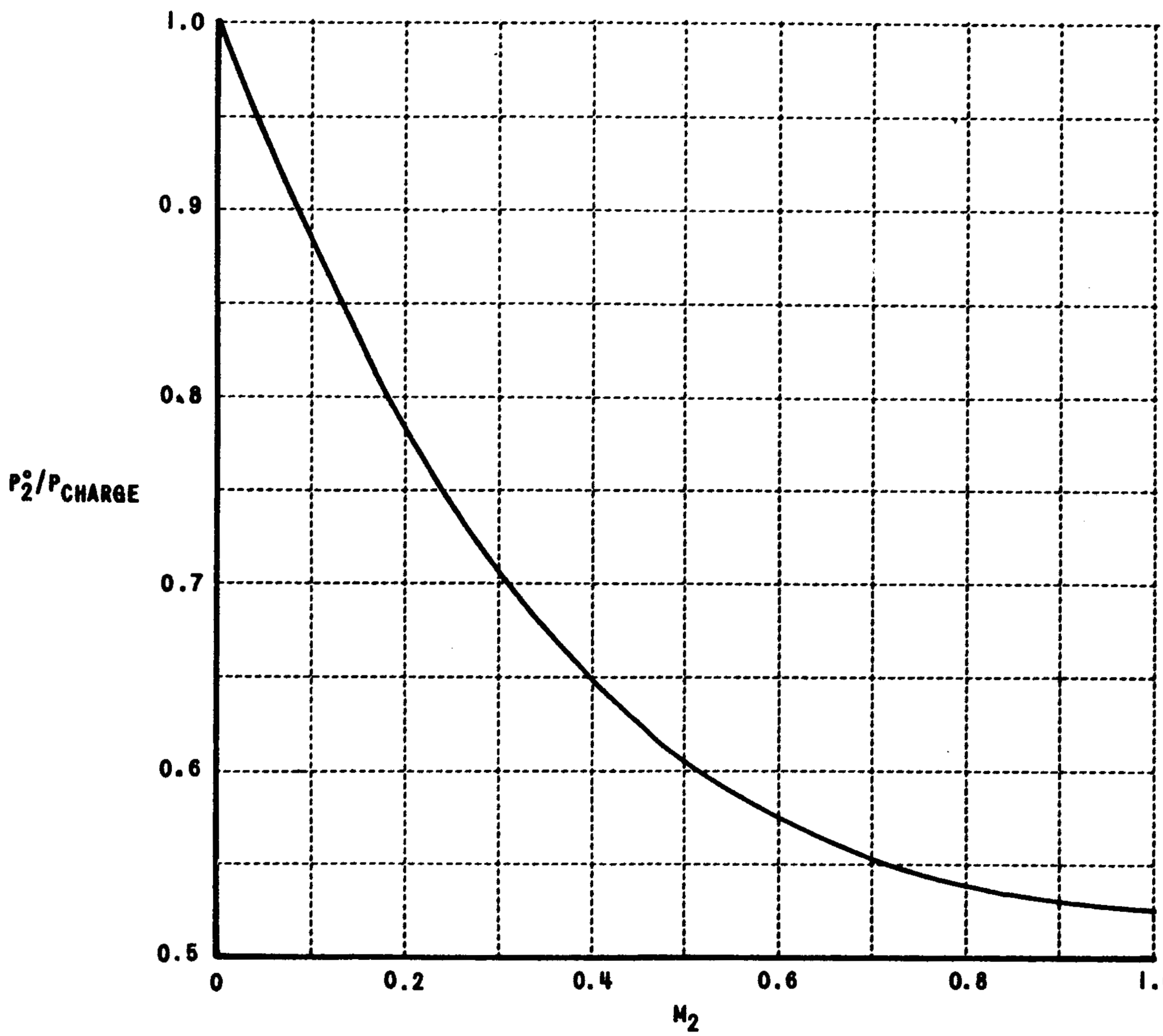


Figure 18 $P_2{}^{o}/P$ CHARGE VS MACH NUMBER IN SUPPLY TUBE

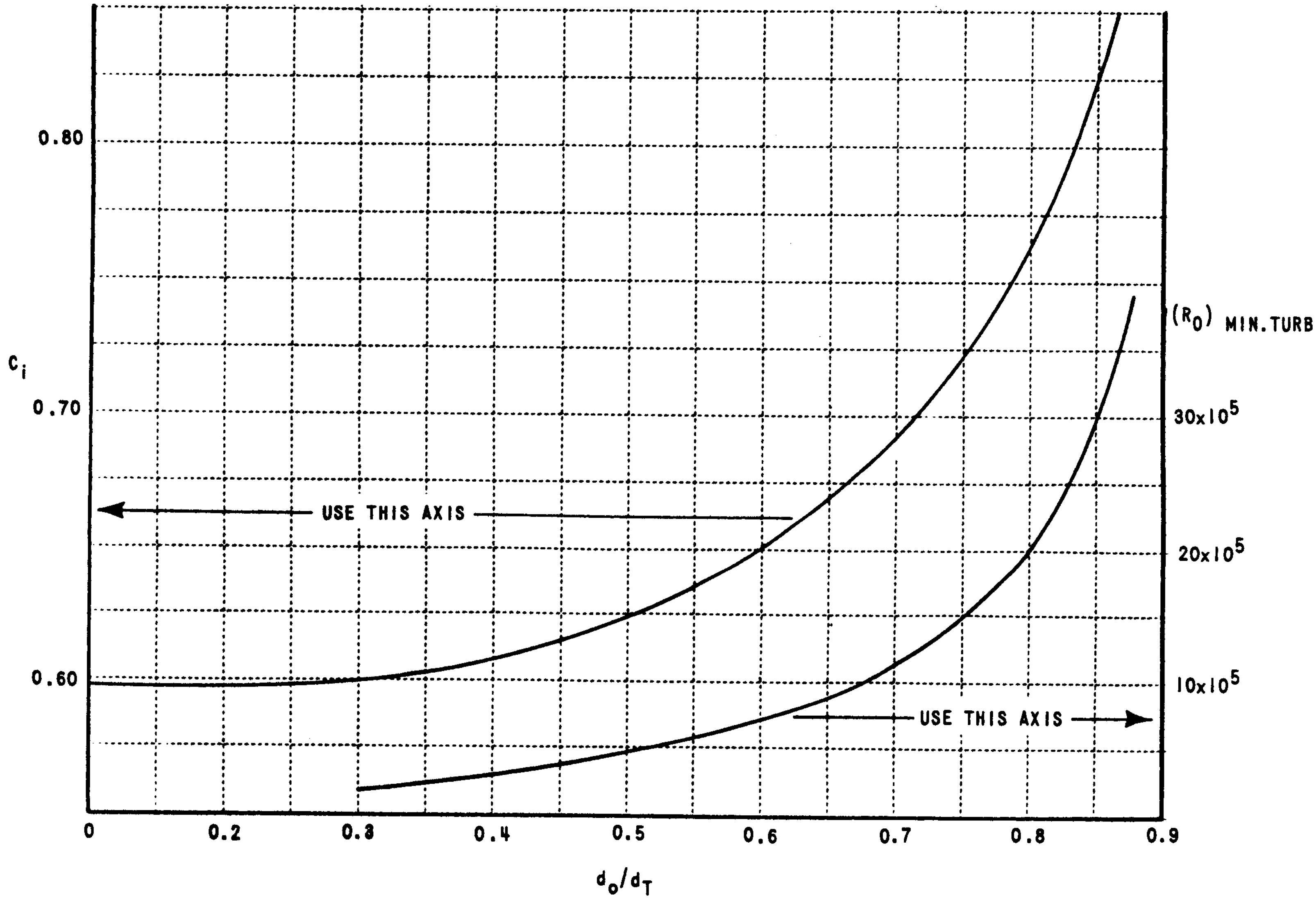


Figure 19 INCOMPRESSIBLE ORIFICE COEFFICIENT AND MINIMUM REYNOLDS NUMBER FOR TURBULENT FLOW VS ORIFICE TO TUBE DIAMETER RATIO

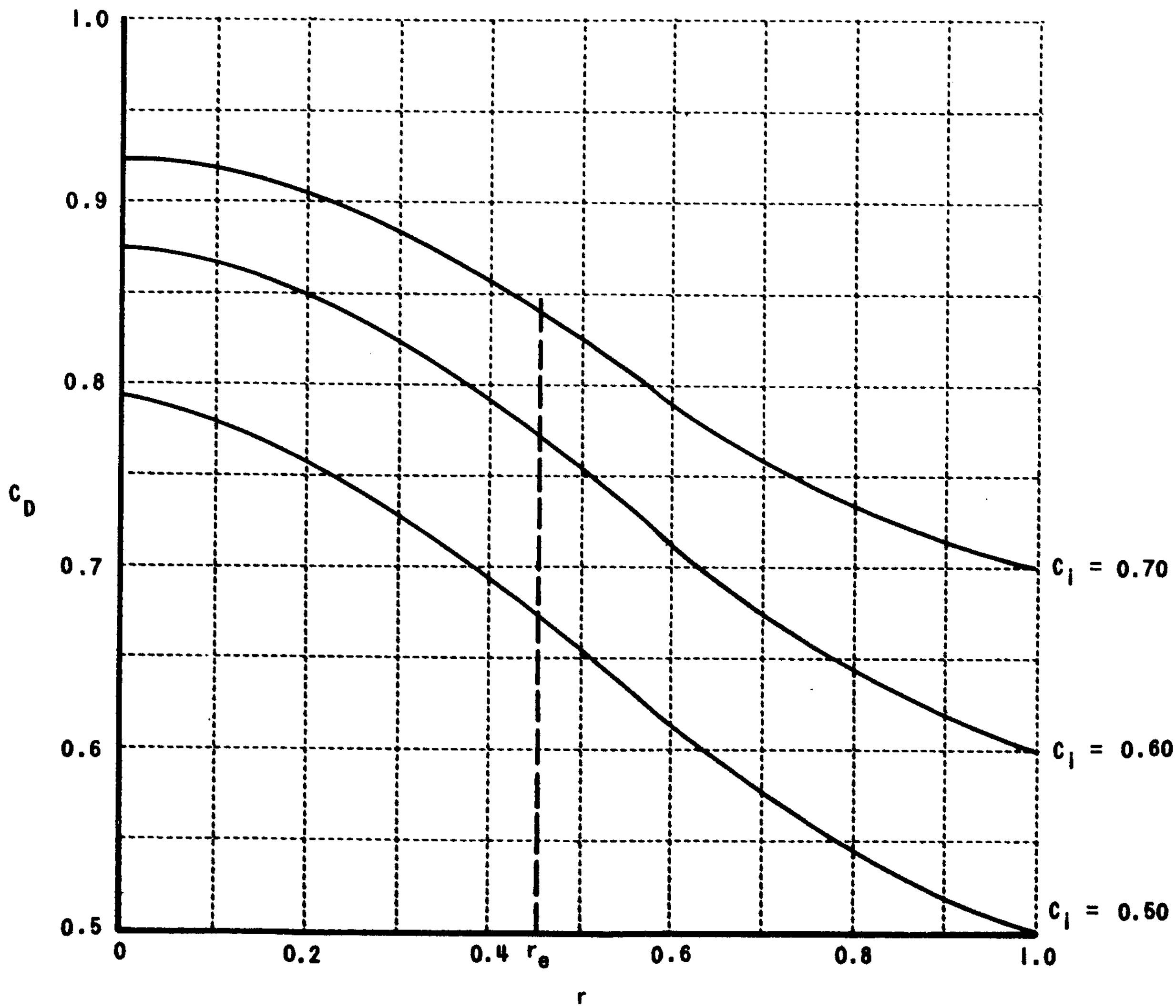


Figure 20 C_D VS r FOR VARIOUS C_i

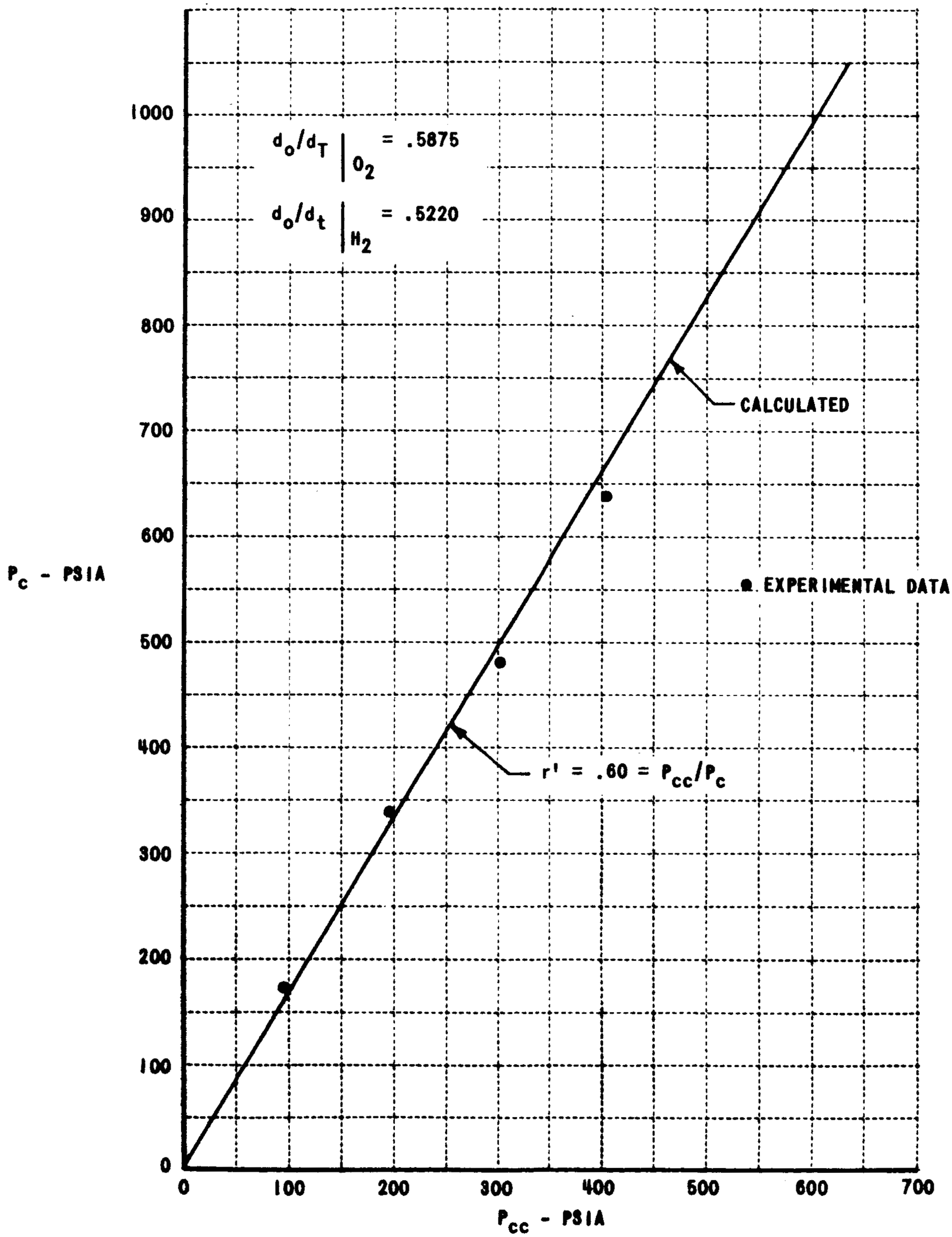


Figure 21 COMPARISON OF CALCULATED VS EXPERIMENTAL COMBUSTOR PERFORMANCE